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JOURNAL OF THE AMERICAN

ROCKET

SOCIETY

A journal devoted to rocket technology and the jet propulsion sciences

VOLUME 23 JULY-AUGUST 1953 NUMBER 4

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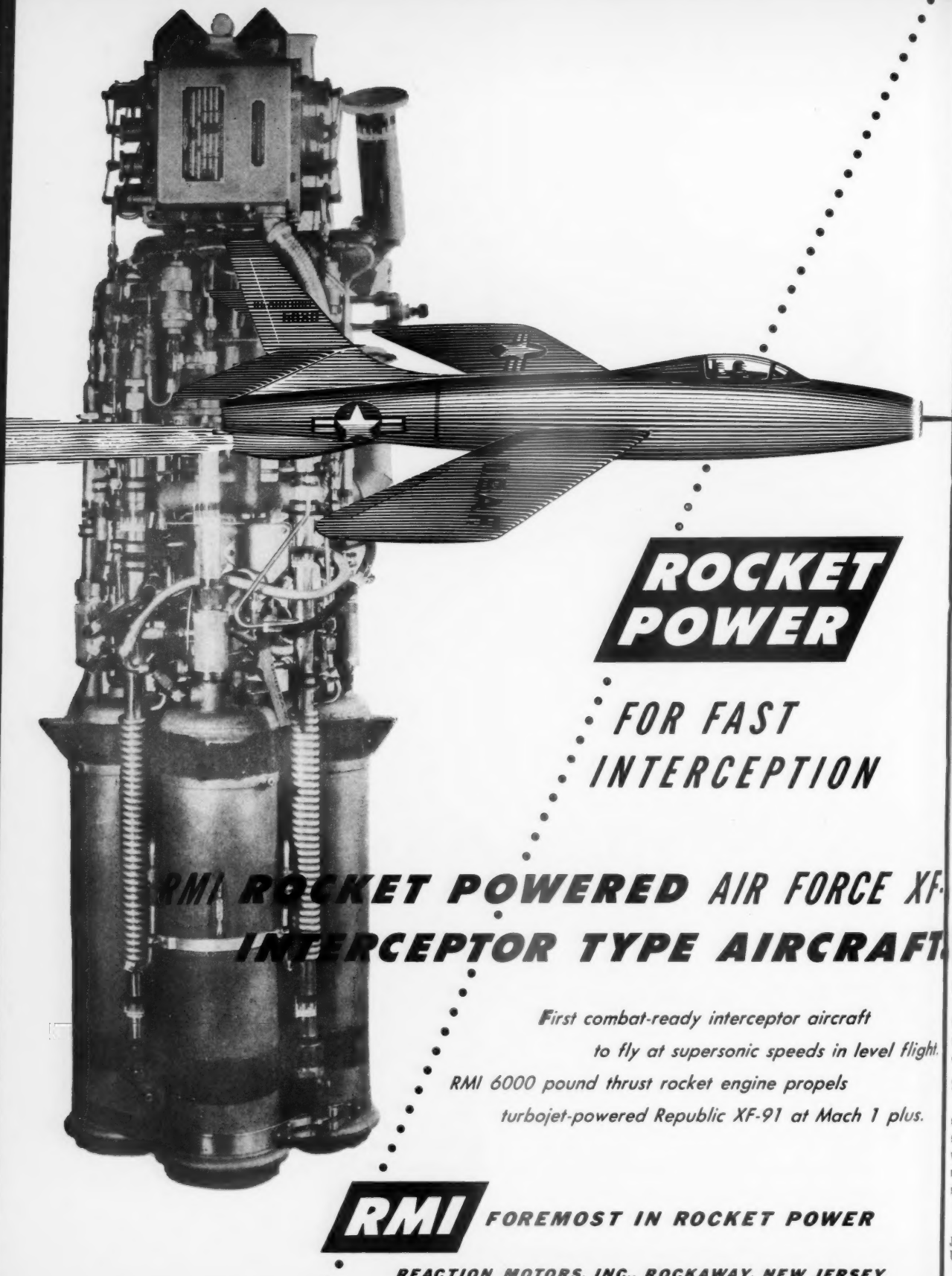

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The Journal of the American Rocket Society is devoted to the advancement of the field of jet propulsion through the publication of original papers disclosing new knowledge and new developments. The term "jet propulsion" as used herein is understood to embrace all engines that develop thrust by rearward discharge of a jet through a nozzle or duct, and thus includes systems utilizing atmospheric air and underwater systems, as well as rocket engines. The Journal is open to contributions, either fundamental or applied, dealing with specialized aspects of jet and rocket propulsion, such as fuels and propellants, combustion, heat transfer, high temperature materials, mechanical design analyses, flight mechanics of jet-propelled vehicles, astronautics, and so forth. The Journal endeavors, also, to keep its subscribers informed of the affairs of the Society and of outstanding events in the rocket and jet propulsion field.

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Represents the jet propulsion profession before the public, promotes interest in jet propulsion and its applications, encourages the formation of regional chapters and student branches where technical conferences may be held.

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Encourages and aids those actively working in this field by organizing national and regional technical meetings where contributed and invited papers are presented and discussed.

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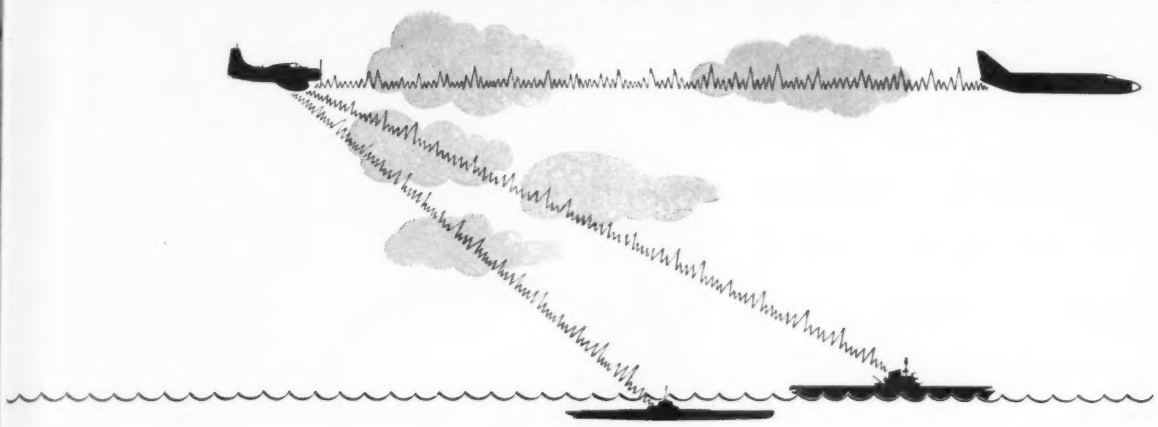

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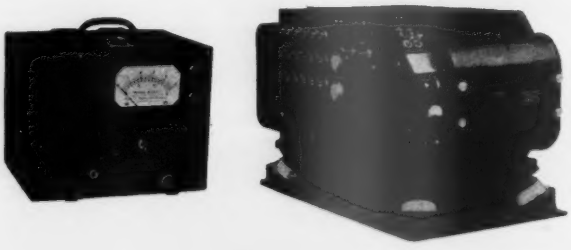

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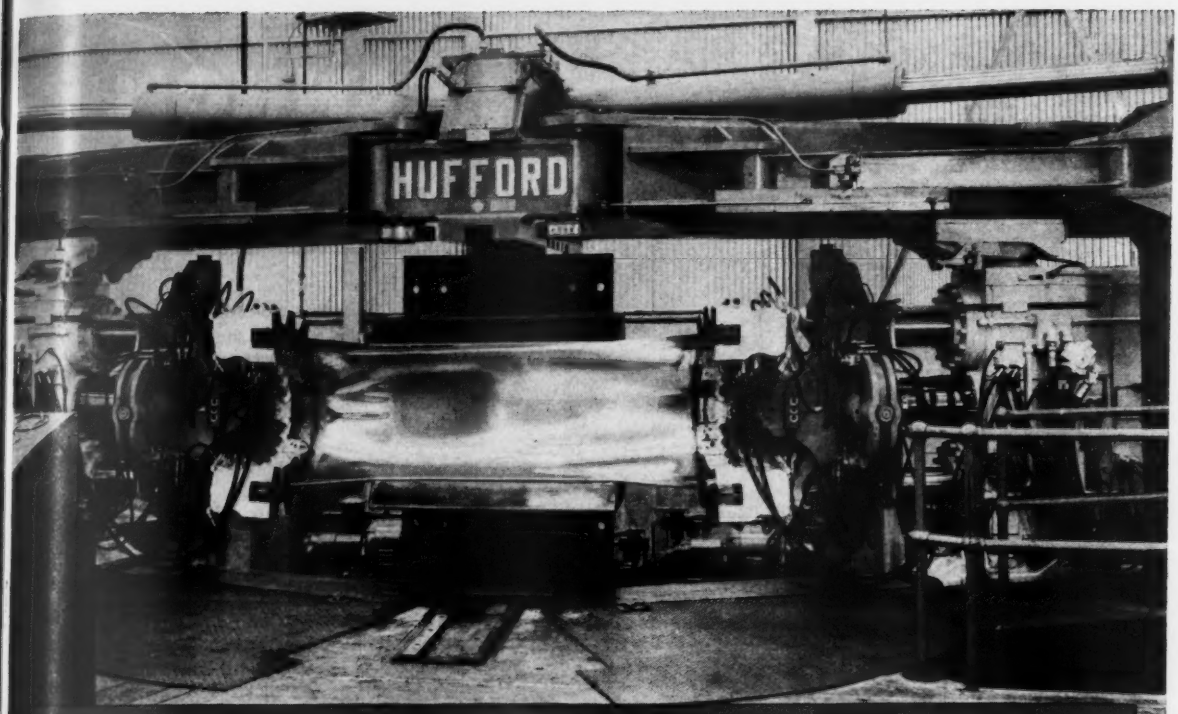

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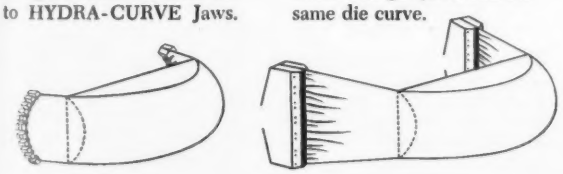

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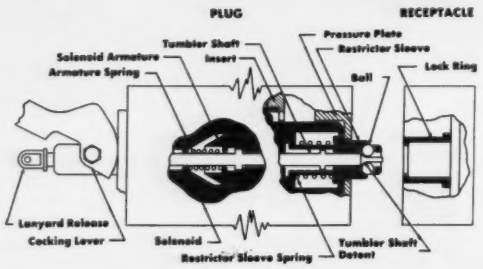

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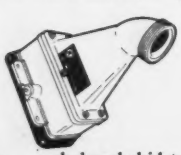

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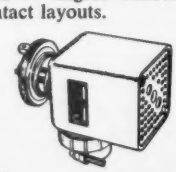

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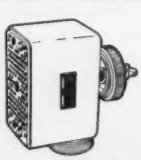

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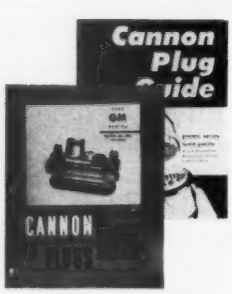

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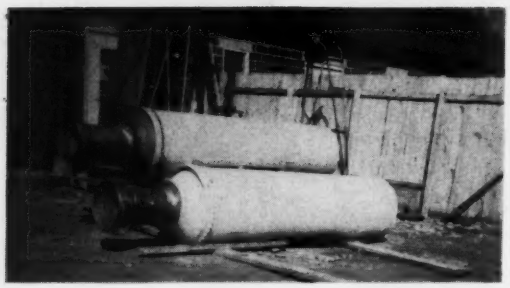

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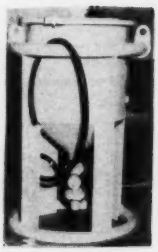

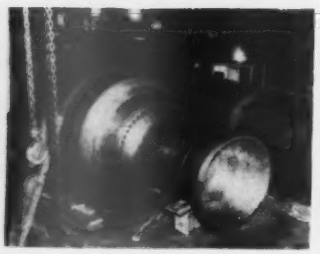

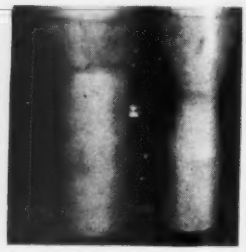

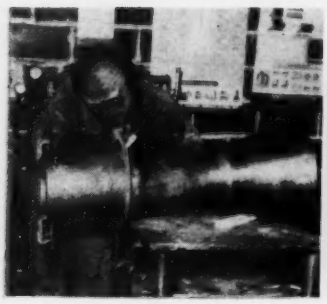

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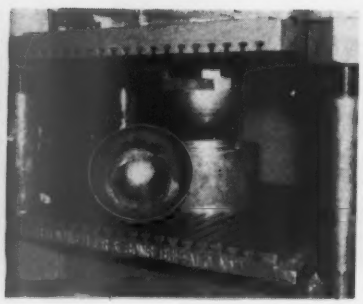

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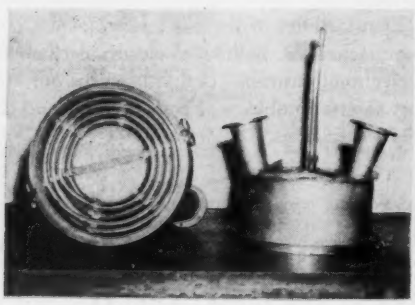

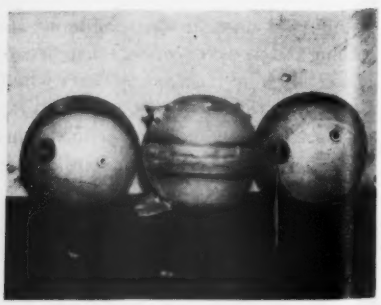

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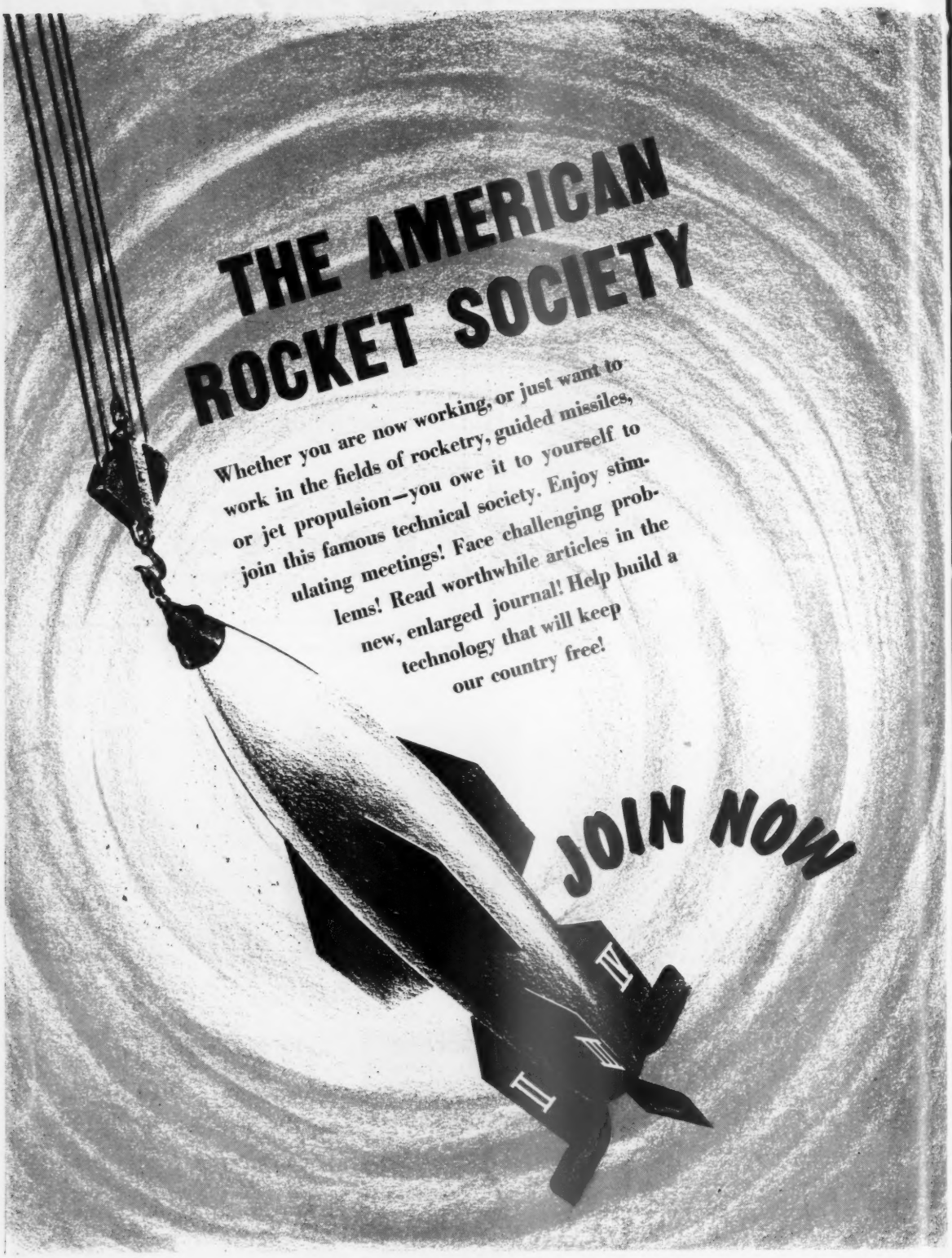
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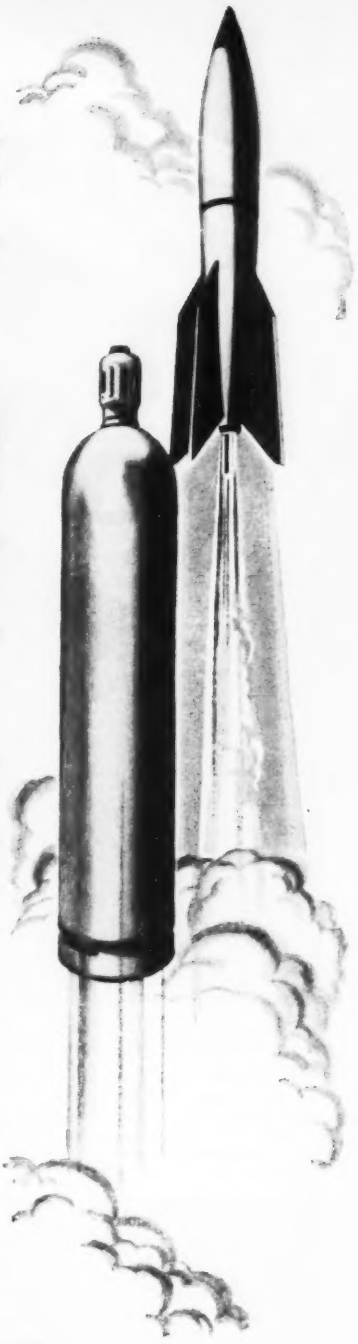

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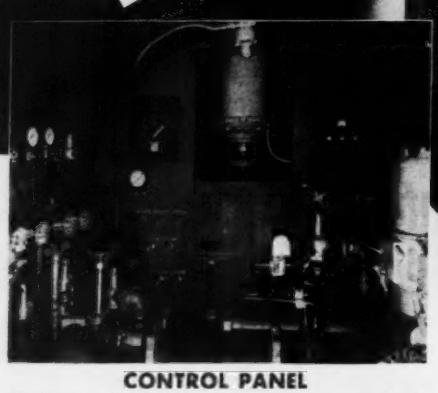
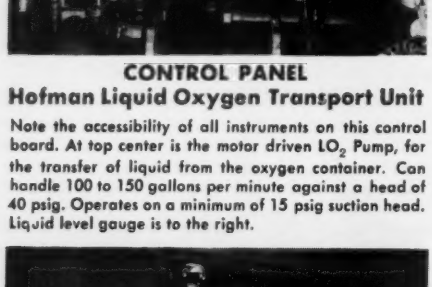


CONTROL PANEL
Hofman Liquid Oxygen Transport Unit

Note the accessibility of all instruments on this control board. At top center is the motor driven LO_2 Pump, for the transfer of liquid from the oxygen container. Can handle 100 to 150 gallons per minute against a head of 40 psig. Operates on a minimum of 15 psig suction head. Liquid level gauge is to the right.



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Editor-in-Chief
MARTIN SUMMERFIELD

Take-Off from Satellite Orbit

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The mass ratio or the characteristic velocity for the take-off of a space ship from the satellite orbit is computed for two cases: the radial thrust, and the circumferential thrust. The circumferential thrust is much more efficient in that the required mass ratio is much less than for the radial thrust. Both cases show, however, an increase of the required mass ratio and the characteristic velocity with a reduction in acceleration. With circumferential thrust, the characteristic velocity increases by a factor of two, when the acceleration is reduced from $^1/_2$ g to $^1/_{3000}$ g.

FOR take-off of a rocket from the earth surface, it is convenient to have the initial trajectory in the vertical direction, and then the thrust should be considerably larger than the initial weight of the rocket to overcome the gravity and to give an appropriate acceleration. Depending upon the relative magnitudes of the aerodynamic drag and the weight, the initial ratio of the thrust and the weight should be between 2 and 3 for minimum expenditure of the propellant. The situation is quite different for a space ship taking off from the satellite orbit: In a satellite orbit, the gravitational attraction is completely balanced by the centrifugal force, and the vehicle is effectively in a weightless state. This fact has led many fanciers of interplanetary travel to conclude that take-off from satellite orbit requires only a very minute thrust. For instance, L. Spitzer (1)[2] proposed a nuclear power plant for a space ship to be accelerated at only $^1/_{3000}$ g. Another example is the extensive discussion of interorbital transport techniques by H. Preston-Thomas (2), based upon the assumption of equally small acceleration. On the other hand, W. von Braun (3) seems to prefer a very much larger acceleration of approximately $^1/_2$ g for take-off from the satellite orbit.

The magnitude of the acceleration has a strong bearing on the optimum type of power plant to be used: The ion-beam rocket is only feasible for very small acceleration, while for moderate acceleration, chemical rocket is required. Therefore the question of the magnitude of acceleration is an important one for interplanetary flight. The purpose of this note is to compute the relation between the acceleration and the mass ratio required for escape from the earth's gravitational field, starting from the satellite orbit. It is hoped that the present investigation will give the future generation of astronautical engineers a rational basis for designing space ships.

Received November 19, 1952.

[1] Robert H. Goddard Professor of Jet Propulsion.

[2] Numbers in parentheses refer to the References on page 236

Basic Equations

The problem considered is the motion of a space ship under the influence of the rocket thrust and the gravitational attraction of a single massive body, say the earth. Then if the rocket thrust is in the plane of trajectory, the trajectory of the space ship will remain in a plane. Let the position of the ship at any time instant t be given by the polar co-ordinates r and θ (r is the distance from the center of attraction, and θ the angular position). If the components of the rocket thrust per unit mass of the vehicle are R in the radial direction and Θ in the circumferential direction, and if g is the magnitude of gravitational attraction at the starting satellite orbit $r = r_0$ (Fig. 1), then the equations of motion of the space ship are

$$\frac{d^2r}{dt^2} = R + r\left(\frac{d\theta}{dt}\right)^2 - g\left(\frac{r_0}{r}\right)^2 \quad \ldots\ldots [1]$$

and

$$\frac{d}{dt}\left(r^2\frac{d\theta}{dt}\right) = r\,\Theta \quad \ldots\ldots [2]$$

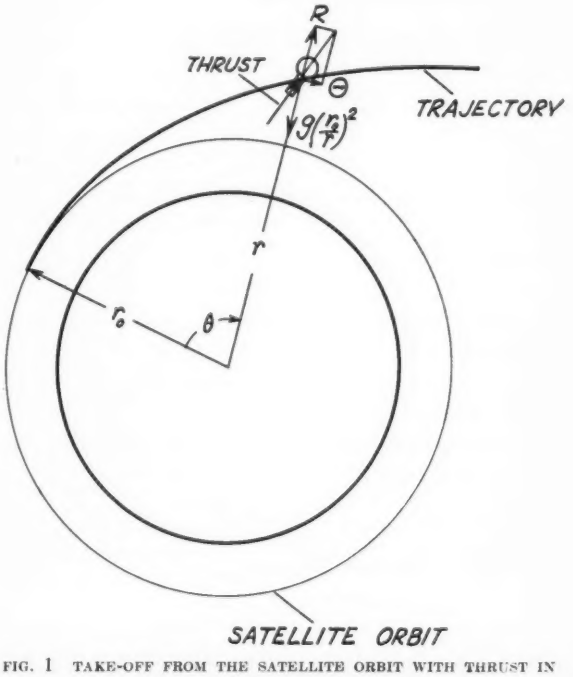

FIG. 1 TAKE-OFF FROM THE SATELLITE ORBIT WITH THRUST IN THE PLANE OF SATELLITE ORBIT

By using the subscript 0 to indicate quantities at the starting instant $t = 0$, the equilibrium condition of the satellite orbit is given by

$$r_0\left(\frac{d\theta}{dt}\right)_0^2 = g \quad \ldots\ldots [3]$$

Initially, the radial velocity is zero, i.e.,

$$\left(\frac{dr}{dt}\right)_0 = 0 \quad \ldots\ldots [4]$$

These are the initial conditions.

For the space ship to have sufficient energy to escape the earth gravitational field at the end of the powered flight, the sum of the kinetic energy and potential energy must vanish at the end of the accelerating period. Let that instant be denoted by the subscript 1. Thus, at $t = t_1$

$$\frac{1}{2}\left[\left(\frac{dr}{dt}\right)_1^2 + \left(r\frac{d\theta}{dt}\right)_1^2\right] - g\frac{r_0^2}{r_1} = 0 \quad \ldots\ldots [5]$$

With any specified variation of the thrust forces R and Θ as functions of time, the above system of equations determine completely the take-off trajectory of the space ship. In the following sections, two special cases of practical significance will be discussed in detail: the case $R = \text{const}$, $\Theta = 0$, purely radial thrust; and the case $R = 0$, $\Theta = \text{const}$, purely circumferential thrust.

Radial Thrust

If the thrust is always radial and is proportional to the instantaneous mass of the vehicle, a nondimensional thrust factor μ can be introduced as

$$R = \mu g \quad \ldots\ldots [6]$$

Furthermore, let

$$\rho = \frac{r}{r_0}, \quad \tau = \sqrt{\frac{g}{r_0}}\,t \quad \ldots\ldots [7]$$

ρ is thus the nondimensional radial distance, and τ is the nondimensional time. Then Equations [1] and [2] can be written in the nondimensional form as

$$\frac{d^2\rho}{d\tau^2} = \mu + \rho\left(\frac{d\theta}{d\tau}\right)^2 - \frac{1}{\rho^2} \quad \ldots\ldots [8]$$

and

$$\frac{d}{d\tau}\left(\rho^2\frac{d\theta}{d\tau}\right) = 0 \quad \ldots\ldots [9]$$

Equation [9] can be immediately integrated and by using the initial condition of Equation [3], the result of integration is

$$\frac{d\theta}{d\tau} = \frac{1}{\rho^2} \quad \ldots\ldots [10]$$

By substituting this equation into Equation [8], the final equation for ρ is

$$\frac{d^2\rho}{d\tau^2} = \mu + \frac{1}{\rho^3} - \frac{1}{\rho^2} \quad \ldots\ldots [11]$$

The nondimensional radial velocity is $d\rho/d\tau$. This is related to the physical radial velocity dr/dt as follows

$$\frac{dr}{dt} = \sqrt{gr_0}\frac{d\rho}{d\tau} \quad \ldots\ldots [12]$$

Equation [11] can be rewritten as

$$\frac{1}{2}\frac{d}{d\rho}\left(\frac{d\rho}{d\tau}\right)^2 = \mu + \frac{1}{\rho^3} - \frac{1}{\rho^2}$$

Since $d\rho/d\tau = 0$, when $\tau = 0$ and $\rho = 1$ according to Equation [4], the result of integrating the above equation is

$$\left(\frac{d\rho}{d\tau}\right)^2 = 2\mu(\rho - 1) + \left(1 - \frac{1}{\rho^2}\right) - 2\left(1 - \frac{1}{\rho}\right) \quad \ldots\ldots [13]$$

Therefore the nondimensional time τ can be calculated as a function of the radius ρ as follows

$$\tau = \int_1^\rho \frac{\rho d\rho}{\sqrt{(\rho - 1)(2\mu\rho^2 - \rho + 1)}} \quad \ldots\ldots [14]$$

With Equations [10] and [13], the end condition of Equation [5] can be written as

$$\frac{1}{2}\left[\left\{2\mu(\rho_1 - 1) + \left(1 - \frac{1}{\rho_1^2}\right) - 2\left(1 - \frac{1}{\rho_1}\right)\right\} + \frac{1}{\rho_1^3}\right] - \frac{1}{\rho_1} = 0$$

Or simply

$$\rho_1 = 1 + \frac{1}{2\mu} \quad \ldots\ldots [15]$$

Then the velocities at the end of acceleration period are

$$\left.\begin{aligned}\left(\frac{dr}{dt}\right)_1 &= \sqrt{gr_0}\,\frac{\sqrt{1 + (1/\mu)}}{1 + (1/2\mu)} \\ \text{and} \\ \left(r\frac{d\theta}{dt}\right)_1 &= \sqrt{gr_0}\,\frac{1}{1 + (1/2\mu)}\end{aligned}\right\} \quad \ldots\ldots [16]$$

The time τ_1 for the powered flight can be obtained from Equation [14] by setting the upper limit of integration to ρ_1. The result of this integration is[3]

$$\tau_1 = \sqrt{\frac{2}{\mu}}\left[\frac{\sqrt{2(\mu + 1)}}{2\mu + 1} + F\left(\frac{1}{\sqrt{8\mu}}, \cos^{-1}\frac{2\mu - 1}{2\mu + 1}\right) + E\left(\frac{1}{\sqrt{8\mu}}, \cos^{-1}\frac{2\mu - 1}{2\mu + 1}\right)\right] \quad \ldots\ldots [17]$$

where F and E are the elliptical integrals of first kind and second kind, respectively.

If $M(t)$ is the instantaneous mass of the space ship, and c the effective exhaust velocity of the rocket, then

$$RM = \mu g M = -c\frac{dM}{dt} = -c\sqrt{\frac{g}{r_0}}\frac{dM}{d\tau}$$

Therefore the mass ratio M_0/M_1 can be calculated as follows:

$$\log_e(M_0/M_1) = \frac{\sqrt{gr_0}}{c}\mu\tau_1$$

By using the result of Equation [17]

$$\frac{c}{\sqrt{gr_0}}\log_e(M_0/M) = \frac{2\sqrt{\mu(\mu + 1)}}{2\mu + 1} + \sqrt{2\mu}\left\{F\left(\frac{1}{\sqrt{8\mu}}, \cos^{-1}\frac{2\mu - 1}{2\mu + 1}\right) + E\left(\frac{1}{\sqrt{8\mu}}, \cos^{-1}\frac{2\mu - 1}{2\mu + 1}\right)\right\} \quad \ldots [18]$$

When the acceleration is very large, $\mu \gg 1$, the integrand in Equation [14] can be expanded in terms of this parameter. Then the mass ratio is calculated as

$$\frac{c}{\sqrt{gr_0}}\log_e(M_0/M_1) = 1 + \frac{1}{24\mu^2} - \frac{1}{40\mu^3} + \ldots\ldots [19]$$

The relation of Equations [18] and [19] is plotted in Fig. 2. For $\mu = {}^1/_8$, the mass ratio becomes infinite. The reason is that at this value of acceleration, there is a radial position where the thrust force is equal to the gravitational attraction and no further increase in the energy of the vehicle can occur. Therefore the radial thrust per unit mass, if maintained constant throughout the powered flight, should be larger than ${}^1/_8$ g. With increasing thrust, the required mass ratio for es-

[3] The author is indebted to Dr. Y. T. Wu who kindly supplied the relation of Equation [17].

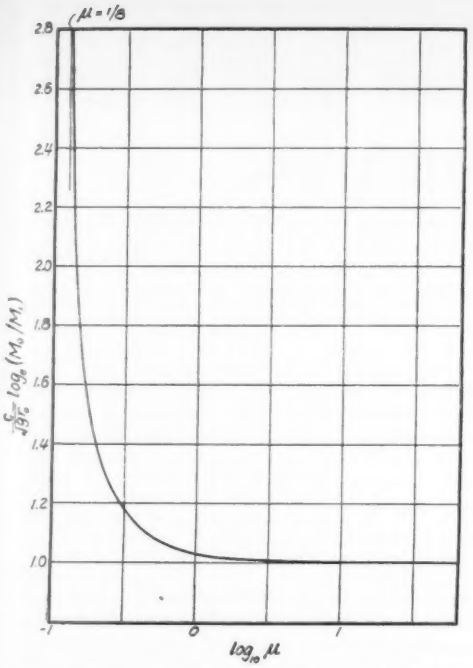

FIG. 2 MASS RATIO FACTOR $(c/\sqrt{gr_0})\ \log_e (M_0/M_1)$ AGAINST ACCELERATION FACTOR μ FOR RADIAL THRUST. c, EFFECTIVE EXHAUST VELOCITY; g, GRAVITY AT THE SATELLITE ORBIT OF RADIUS r_0; M_0, INITIAL MASS; M_1, FINAL MASS; μ, THE RATIO OF INSTANTANEOUS THRUST PER UNIT MASS AND g FOR RADIAL THRUST

cape from the earth's gravitational field decreases. This strong dependence of the mass ratio upon the acceleration factor is contrary to opinion that for take-off from satellite orbit only very small thrust is required. The asymptotic value of $\log_e (M_0/M_1)$ is $\sqrt{gr_0}/c$. However, there is no appreciable improvement in going to higher thrust than 1 g.

Equation [16] shows that at very large values of the acceleration factor μ, the acceleration is accomplished in so short an interval that the circumferential velocity at the end of the acceleration remains at the initial value of $\sqrt{gr_0}$. The radial velocity increases from nothing at the initial instant to the final value of $\sqrt{gr_0}$. The total kinetic energy is thus gr_0 at the end of acceleration and this is equal to the negative of potential energy at that instant, since the radial position r must be practically the initial value r_0 under very large thrust. The work of the rocket is to produce the radial velocity $\sqrt{gr_0}$. Thus it is evident that the value of $c \log_e (M_0/M_1)$ must be $\sqrt{gr_0}$, as the calculation shows.

Circumferential Thrust

If the thrust is always circumferential and proportional to the mass of the vehicle, then a new thrust factor ν can be introduced such that

$$\Theta = \nu g \qquad [20]$$

By using the same nondimensional variables as defined in Equation [6], the equations of motion are

$$\frac{d^2\rho}{d\tau^2} = \rho \left(\frac{d\theta}{d\tau}\right)^2 - \frac{1}{\rho^2} \qquad [21]$$

$$\frac{d}{d\tau}\left(\rho^2 \frac{d\theta}{d\tau}\right) = \nu\rho \qquad [22]$$

The initial conditions of Equation [3] and [4] are

$$\left(\frac{d\theta}{d\tau}\right)_0 = 1, \quad \left(\frac{d\rho}{d\tau}\right)_0 = 0 \quad \text{at} \quad \rho = 1, \tau = 0 \qquad [23]$$

Therefore, Equation [21] gives another initial condition that

$$\left(\frac{d^2\rho}{d\tau^2}\right)_0 = 0 \qquad [24]$$

By eliminating θ from Equations [21] and [22]

$$\frac{d}{d\tau}\left(\rho^3 \frac{d^2\rho}{d\tau^2} + \rho\right)^{1/2} = \nu\rho \qquad [25]$$

This is a third-order differential equation with three initia conditions specified by Equations [23] and [24]. No simple general solution can, however, be obtained. The following discussion will be concerned with the approximations that are valid for large values of ν or for small values of ν.

For very large values of ν, the acceleration period is expected to be short and the change of the radial position to be small. Then the value of ρ must be very close to the initial value of unity. By taking ρ to be unity, Equation [25] becomes

$$\frac{d}{d\tau}\left(\frac{d^2\rho}{d\tau^2} + 1\right)^{1/2} = \nu$$

Then

$$\frac{d^2\rho}{d\tau^2} + 1 = C^2 + 2C\nu\tau + \nu^2\tau^2$$

where C is the integration constant. C, however, must be 1 because of the initial condition of Equation [24]. The appropriate approximate solution for ρ for very large ν is thus

$$\rho \cong 1 + \frac{1}{3}\nu\tau^3 + \frac{1}{12}\nu^2\tau^4 \qquad [26]$$

To obtain higher terms in this power series, the usual series substitution method may be used. The calculation is somewhat lengthy and therefore will not be reproduced there. The result is

$$\rho = 1 + \frac{1}{3}\nu\tau^3 + \frac{1}{12}\nu^2\tau^4 - \frac{\nu}{60}\tau^5 - \frac{23\nu^2}{360}\tau^6 + \ldots \qquad [27]$$

By using the result of Equation [27], the radial velocity is obtained by differentiation. Then Equation [21] gives the circumferential velocity. The end condition of Equation [5] can be modified into the following more convenient form by mulitiplying it by $2r^2$

$$0 = \left[\left(\rho \frac{d\rho}{d\tau}\right)^2 + \left(\rho^2 \frac{d\theta}{d\tau}\right)^2 - 2\rho\right]$$

By substituting the solution of Equation [27] into this condition, an equation for determining τ_1 is obtained

$$0 = -1 + 2\nu\tau_1 + \nu^2\tau_1^2 - \frac{2}{3}\nu\tau_1^3 + \nu^2\tau_1^4 + \frac{\nu}{30}(1 + 26\nu^2)\tau_1^5 - \frac{\nu^2}{90}(4 - 13\nu^2)\tau_1^6 + \ldots \qquad [28]$$

The mass ratio M_0/M_1 can be calculated in the same way as in the previous section and can be determined through the new parameter x defined as follows

$$\frac{c}{\sqrt{gr_0}} \log_e (M_0/M_1) = \nu\tau_1 = x \qquad [29]$$

Equation [28] then can be written as

$$0 = -1 + 2x + x^2 - \frac{2}{3}\frac{x^3}{\nu^2} + \frac{x^4}{\nu^2} + \frac{x^5}{30\nu^4} + \frac{13}{15}\frac{x^5}{\nu^2} - \frac{2}{45}\frac{x^6}{\nu^4} + \frac{13}{90}\frac{x^6}{\nu^2} + \ldots \qquad [30]$$

Since the calculation is designed for large values of ν, the appropriate expansion of x should be a series in inverse powers ν. Equation [30] suggests specifically

$$x(\nu) = x^{(0)} + \frac{x^{(1)}}{\nu^2} + \frac{x^{(2)}}{\nu^4} + \ldots \qquad [31]$$

where $x^{(0)}$, $x^{(1)}$, and $x^{(2)}$ are constants independent of ν. By

substituting Equation [31] into Equation [30] and equating equal powers of ν, the following set of equations results.

$$x^{(0)2} + 2x^{(0)} - 1 = 0 \quad \ldots [32]$$

$$x^{(1)} = \frac{1}{2(1+x^{(0)})}\left[\frac{2}{3}x^{(0)3} - x^{(0)4} - \frac{13}{15}x^{(0)5} - \frac{13}{90}x^{(0)6}\right] \quad \ldots [33]$$

$$x^{(2)} = \frac{1}{2(1+x^{(0)})}\left[-x^{(1)2} + 2x^{(0)2}x^{(1)} - 4x^{(0)3}x^{(1)} - \frac{1}{30}x^{(0)6} - \frac{13}{3}x^{(0)4}x^{(1)} + \frac{2}{45}x^{(0)6} - \frac{13}{15}x^{(0)5}x^{(1)}\right] \quad \ldots [34]$$

The explicit numerical solutions are then

$$\begin{aligned} x^{(0)} &= \sqrt{2} - 1 = 0.41421 \\ x^{(1)} &= 0.002349 \\ x^{(2)} &= -0.00004791 \end{aligned} \quad \ldots [35]$$

This completes the calculation of mass ratio for large values of the acceleration factor ν.

For the other extreme case of very small values of ν, it is to be expected that the acceleration will be very small, and in Equation [25] the term $\rho^3 d^2\rho/d\tau^2$ will be very much smaller than ρ. Therefore a good approximation of Equation [25] at small ν is

$$\frac{d}{d\tau}\rho^{1/2} = \nu\rho \qquad \text{or} \qquad \frac{1}{2}\frac{d\rho}{\rho^{3/2}} = \nu\, d\tau$$

The solution of this equation with the initial condition of $\rho = 1$ at $\tau = 0$ is

$$\rho = \frac{1}{(1-\nu\tau)^2} \quad \ldots [36]$$

Therefore

$$\frac{d\rho}{d\tau} = \frac{2\nu}{(1-\nu\tau)^3}, \quad \frac{d^2\rho}{d\tau^2} = \frac{6\nu^2}{(1-\nu\tau)^4} \quad \ldots [37]$$

At $\tau = 0$, the radial velocity and the radial acceleration are thus not zero, as required by the initial conditions of Equations [23] and [24]. They are, however, very small, because ν is very small. Therefore the solution of Equation [36] is a good approximation to the exact solution.

To the same approximation, Equation [20] becomes

$$\rho\frac{d\theta}{d\tau} = \frac{1}{\rho^{1/2}} = (1-\nu\tau) \quad \ldots [38]$$

This means that at every instant, because of the extremely small acceleration, the centrifugal force per unit mass $r\,(d\theta/dt)^2$ practically balances the gravitational attraction. The end condition of Equation [5] can then be written as

$$\frac{4\nu^2}{(1-x)^6} - (1-x)^2 = 0 \quad \ldots [39]$$

where x is again $\nu\tau_1$. The appropriate solution for x is then

$$x = 1 - (2\nu)^{1/4} \quad \ldots [40]$$

Since the mass ratio, M_0/M_1, is related to x by Equation [29], Equation [40] actually gives the mass ratio for escaping the gravitational field with very small acceleration.

The parameter x is plotted against ν in Fig. 3, using Equation [31] with both Equations [35] and [40]. When ν approaches zero, x approaches 1. When ν is very large, x approaches $\sqrt{2}-1$. As ν increases, x and hence the mass ratio, M_0/M_1, decrease monotonically. Therefore, same as the result for purely radial thrust, there is a strong influence of the magnitude of acceleration on the required mass ratio. However, as far as decreasing the mass ratio is concerned, there is no appreciable advantage in using ν greater than $1/2$.

When the acceleration factor ν is very large, the thrust force acts like an impulse. Since the thrust is in the circumferential direction, the rocket action only produces an increase in the circumferential velocity with practically no change in the radial position. The initial circumferential velocity is $\sqrt{gr_0}$ the required circumferential velocity for escape is $\sqrt{2gr_0}$. Thus the increase of velocity produced by the rocket action is $(\sqrt{2}-1)\sqrt{gr_0}$. This explains the asymptotic value of x for very large ν.

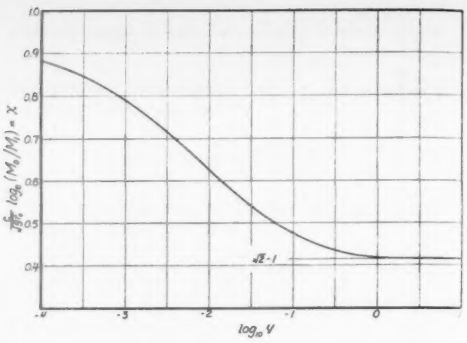

FIG. 3 MASS RATIO FACTOR $(c/\sqrt{gr_0})\ \log_e\,(M_0/M_1)$ AGAINST ACCELERATION FACTOR ν FOR CIRCUMFERENTIAL THRUST. c, EFFECTIVE EXHAUST VELOCITY; g, GRAVITY AT THE SATELLITE ORBIT OF RADIUS r_0; M_0 INITIAL MASS; M_1, FINAL MASS; ν, THE RATIO OF INSTANTANEOUS THRUST PER UNIT MASS AND g FOR CIRCUMFERENTIAL THRUST

Discussion

By comparing Fig. 2 with Fig. 3, it is apparent that the radial thrust is much less efficient than the circumferential thrust for take-off from the satellite orbit. For large thrusts, the value of $\log\,(M_0/M_1)$ for radial thrust is more than twice that for circumferential thrust. Furthermore, in case of radial thrust, the ratio of thrust to the instantaneous mass, if maintained constant, must be larger than $g/8$. In case of circumferential thrust, no such limit exists. Therefore, circumferential thrust is definitely preferred.

The quantity $c\,\log_e\,(M_0/M_1)$ is a measure of the performance or the capability of the vehicle. It has the dimension of a velocity and is actually the increase of velocity which the vehicle is capable of in a space without gravitation. This quantity is conveniently called the *characteristic velocity* of the vehicle. Let this be denoted by V Then for the case of circumferential thrust, Equation [29] gives

$$V = c\,\log_e\,(M_0/M_1) = \sqrt{gr_0}\,x = \frac{S}{\sqrt{2\lambda}}x \quad \ldots [41]$$

where S is the "escape velocity" from the surface of the earth, and λ is the ratio of the radii of the satellite orbit and the earth. S is equal to 11.2 km/sec. Fig. 3 then shows that by decreasing the acceleration from $1/2$ to $1/3000\ g$, x, hence the required characteristic velocity V, will increase by a factor of two. This is a very important point for the designers of space ships.

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Theory of Steady Flow with Mass Addition Applied to Solid Propellant Rocket Motors

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Solution of the problem of steady flow of a perfect gas in a straight channel with mass addition is discussed, and a typical solution is demonstrated for the case of mass addition at constant stagnation enthalpy. This problem is then combined with the discharge characteristics of sonic nozzles, and the solution of the combined problem is demonstrated, using a burning propellant as a mechanism of mass addition at constant enthalpy. It is shown that, when the burning characteristics of the propellant and the thermodynamic properties of the gas are established, the flow field depends upon two dimensionless geometrical variables familiar in the internal ballistics of rocket motors. The discussion places particular emphasis on identity and dependence of ballistic variables and the nature of the problem posed by the solid propellant rocket motor.

Nomenclature

a = Velocity of sound, ft/sec
A_p = Flow channel area in the motor, ft²
A_t = Nozzle throat area, ft²
C = Factor in burning rate rule, dependent on propellant temperature, $lb^{-n}\ ft^{2n+1}\ sec^{-1}$
C_D = Discharge coefficient of nozzle, slug/lb sec
C_p = Specific heat at constant pressure, ft²/sec² °R (= 0.001285 Btu/slug °R)
f = Factor in the rule for burning rate, function of θ, Equation [14]
g_1 = Pressure factor in the burning rate rule
g_2 = Velocity factor in the burning rate rule
h = Average specific enthalpy of propellant gas at a distance x from the front end of the motor, ft²/sec²
J = Ratio of nozzle throat area to internal channel area in a motor
k = Erosion (velocity) factor in burning rate rule, sec/ft
l = Total length of propellant charge, ft
m = Mass flux in the internal flow channel at the point x, slug/sec
m_b = Mass rate of burning of the propellant charge, slug/sec
m_d = Mass rate of discharge of the nozzle, slug/sec
n = Pressure exponent in the rule for burning rate
p = Static pressure at the point x, lb/ft²
p_0 = Static pressure at the point $x = 0$, lb/ft²
p_s = Stagnation pressure at the point x, lb/ft²
q = Perimeter of the charge burning surface in a plane at right angles to the charge axis, ft
r = Linear burning rate of the propellant normal to the surface, ft/sec
R = Universal gas constant for perfect gas law, 4.97×10^4 ft lb/slug-mole °R
T = Average static temperature of the stream at the point x, °R
u = Average velocity of the stream at the point x, ft/sec
x = Distance along flow channel from front end of charge, ft
β = Charge geometry parameter proportional to x, $slug^{1-n}\ ft^{n-2}\ sec^{2n-1}$, Equation [15]
γ = Ratio of average specific heats for propellant gas C_p/C_v
θ = Mass flux density parameter, $\theta = m/A_p p_0$, slug/lb sec
μ = Average molecular weight of the products of reaction of the solid propellant
ρ = Average static density of the stream at the point x, slug/ft³
ρ_p = Density of the solid propellant, slug/ft³
φ = Charge geometry parameter, proportional to x, slug/lb sec

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Introduction

THE theory of internal ballistics of solid propellant rocket motors is based largely on a one-dimensional, steady-state theory of flow of a perfect gas in channels (1).[2] This latter theory has been studied with considerably increased intensity in the past 12 years, and a good qualitative understanding of the flow phenomena has been provided by the work of Shapiro and Hawthorne (2) in their summary study of the differential equations for steady, one-dimensional flow. However, even in this drastically restricted realm of hydrothermodynamics, only a few of the problems have been solved to the extent of performing the integration of the differential equations. In these instances many further restrictive assumptions are required. Examples are the case of isentropic flow in a channel of variable cross-sectional area (wind tunnel or rocket nozzle) (3, 4); flow in a channel of constant cross section with heat addition, friction, and chemical reaction (5); flow in a channel of constant cross section with mass addition at constant enthalpy and composition (6); flow in a channel of constant cross section with mass addition at constant enthalpy with chemical reaction (7).

The gas flow in a solid propellant rocket motor can usually be described with respectable accuracy by a channel with two adjoining regions, the upstream one (combustion chamber) corresponding to a constant area channel with mass addition at constant specific enthalpy and zero kinetic energy, the downstream one (nozzle) corresponding to a channel with nonuniform area and isentropic flow. This representation of the flow problem does not appear to have been adequately described in the following respects: (a) The role of the mechanism of mass addition in determining the nature of the problem has not been described as such; (b) the method of solution of the problem of flow with mass addition has not been described in terms of the mechanism of mass addition; and (c) the interaction of the two adjoining flow regions has not been examined as such. A clarification of these points should contribute materially to understanding of rocket motor behavior and prediction of optimum design, and render the results of the theory more accessible to workers in rocket development and evaluation.

In the present paper the rocket motor problem treated by Wimpress is presented with some extension, generalization, and interpretation, with particular emphasis on the aspects of the aforesaid problem. The equations of flow for a rocket motor were presented previously in the present notation (7), using the assumption of steady-state, one-dimensional, adia-

[2] Numbers in parentheses refer to the References on page 241.

batic flow of a perfect gas. Allowance was made for changes in molecular weight and specific entropy due to shift in chemical equilibrium of the reaction products of the propellant. For a large portion of the propellants in use at present, the equilibrium composition of the gas does not change appreciably over the range of pressures and temperatures encountered in rocket motors, and it is satisfactory to assume that the molecular weight and ratio of specific heats are constant, and that the specific enthalpy is a simple function of temperature alone. Under these conditions the solution of the flow problem is much simplified both in method of computation and in form of expression of solution, and permits a clearer insight into the importance of the various parameters of charge geometry and propellant properties.

Statement of the Problem

The arrangement of the components in a typical solid propellant rocket motor is indicated in Fig. 1. During operation, gas flow originates at the forward portion of the propellant charge, and mass is added to the stream everywhere along the channel by burning of the charge. This mass enters the stream with a specific enthalpy characteristic of the solid propellant and virtually independent of the state of the gas

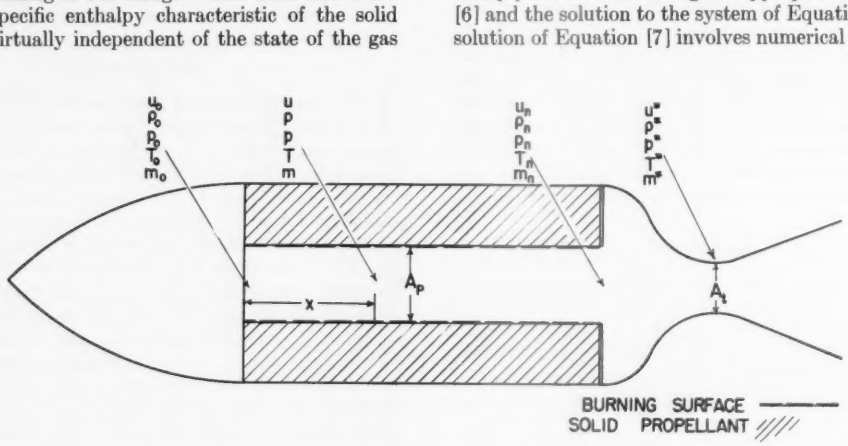

FIG. 1. ARRANGEMENT OF COMPONENTS IN A SOLID PROPELLANT ROCKET MOTOR

stream. As mass is added, the velocity increases, and the pressure drops, with a corresponding decrease in temperature. The actual rate of burning of the propellant depends upon the state of the adjacent stream (8), so that the burning process and the flow field interact on each other in a rather complicated way.

If one assumes that the geometry of the charge is not affected by the forces exerted on it, that the cross-sectional shape of the charge is the same over the entire length of the charge, and that the volume change due to the recession of the burning surface does not affect the momentary state of the flow field, the equations of flow, state, and burning are those of (7). In the present case, it is assumed that the molecular weight and ratio of specific heats of the gas are constant, and that the specific enthalpy is approximated in the region of interest by Equation [5] (9). Under the foregoing conditions, the flow equations are

state	$p = \rho RT/\mu$	[1]
continuity	$m/A_p = \rho u$	[2]
momentum	$p = p_0 - \rho u^2$	[3]
energy	$h + (1/2)u^2 = h_0$	[4]
heat capacity	$h = h(T) = \int C_p dT$	[5]

The foregoing set of five equations has the six variables: p (pressure), ρ (density), T (temperature), m/A_p (mass flux density), u (stream velocity), h (specific enthalpy); and a solution is possible in terms of one of the variables without any consideration of motor or charge geometry. This solution depends only upon the characteristics of the propellant gas expressed by the values of R, μ, h_0, and $h(T)$, and hence need not be repeated for each different mass addition mechanism. A method of solution of Equations [1]–[5] will be outlined in the next section, with a solution for typical values of R, μ, h_0, and $h(T)$.

The solutions of Equations [1]–[5] are related to a particular propellant charge by (7), the two Equations [6] and [7] describing the burning rate of the propellant and the continuity at the burning surface.

$$r = C(T_p)g_1(p)g_2(u) = C_p{}^n(1 + ku) \quad [6]$$

$$m_b = \rho_p q \int_0^x r dx \quad [7]$$

These two equations reflect the interaction between the flow field and the burning of the propellant and introduce two additional variables, namely, the burning rate of the propellant r and the position along the axis of the charge x. Equation [7] can be solved using the appropriate form of Equation [6] and the solution to the system of Equations [1]–[5]. The solution of Equation [7] involves numerical solution of a convenient first-order integral equation. The procedure is outlined in the next section and a sample computation is presented. The solution amounts to determination of the relation between m of Equations [1]–[5] and x of Equation [7] where ρ_p, q, C, n, and k are parameters characteristic of propellant and of charge geometry. This relation between m and x provides the distribution of the other stream variables relative to x through the solution of Equations [1]–[5].

While Equations [1]–[7] are sufficient to describe all one-dimensional steady-state flow fields possible in the present problem, they do not determine which flow field is pertinent in a given motor. This is to be expected, since the field in the motor usually depends upon the size of the nozzle, and no mention has been made of the nozzle size in Equations [1]–[7]. For steady-state operation, the mass flux through the nozzle must equal the mass flux past the downstream end of the charge. The flux at the end of the charge is given by the solution to Equations [1]–[7], and the flux through the nozzle is given by the relation (10)

$$m_d = C_D A_t p_{sn} \quad [8]$$

The pressure p_{sn} is the stagnation pressure at the section between the charge and the nozzle, and is given by the values of p, u, and T at the downstream end of the charge and the relation (11)

$$p_s = p[1 + (\gamma - 1)\mu u^2/2\gamma RT]^{\gamma/(\gamma - 1)} \quad [9]$$

Thus choice of the appropriate solution of Equations [1]–[5]

TABLE 1 SOLUTION OF THE FLOW EQUATIONS FOR HYPOTHETICAL PROPELLANT

1	2	3	4	5	6	7	8	9	10	11	12
$u \times 10^{-3}$, ft/sec	$u^2 \times 10^{-6}$, ft²/sec²	$h \times 10^{-6}$, ft²/sec²	T, °R	$(u^2/T) \times 10^{-2}$, ft²/sec², °R	p/p_0	$(\rho/p_0) \times 10^5$ slug/ft lb	$\theta \times 10^5$ slug/lb sec	p_s/p_0	$JC_D \times 10^5$ slug/lb sec	$f_1(\theta)$	$f_2(\theta)$
0.00	0.000	37.60	4171	0.00	1.000	11.10	0.00	1.000	0.00	1.000	1.000
0.25	0.062	37.57	4169	0.15	0.993	11.03	2.76	0.998	2.76	1.132	1.158
0.50	0.250	37.48	4161	0.60	0.973	10.83	5.41	0.998	5.48	1.256	1.305
0.75	0.562	37.32	4145	1.36	0.941	10.51	7.88	0.972	8.11	1.365	1.438
1.00	1.000	37.10	4128	2.42	0.899	10.09	10.09	0.953	10.59	1.460	1.557
1.25	1.562	36.82	4103	3.81	0.850	9.59	11.99	0.931	12.88	1.543	1.658
1.50	2.250	36.48	4074	5.52	0.796	9.05	13.58	0.908	14.94	1.608	1.743
1.75	3.062	36.07	4038	7.58	0.740	8.49	14.85	0.885	16.77	1.660	1.812
2.00	4.000	35.60	3997	10.01	0.683	7.92	15.83	0.864	18.31	1.703	1.864
2.25	5.062	35.07	3951	12.81	0.628	7.36	16.55	0.847	19.55	1.733	1.905
2.50	6.250	34.48	3900	16.03	0.574	6.81	17.04	0.832	20.47	1.750	1.933
2.75	7.562	33.82	3842	19.68	0.523	6.30	17.34	0.821	21.11	1.757	1.953
3.00	9.000	33.10	3780	23.81	0.476	5.82	17.48	0.819	21.33	1.760	1.957
3.05	9.302	32.95	3767	24.69	0.467	5.74	17.49	0.818	21.39	1.760	1.961
3.10	9.610	32.80	3754	25.60	0.458	5.65	17.51	0.818	21.41	1.760	1.960
3.15	9.922	32.64	3740	26.53	0.449	5.56	17.50	0.818	21.41	1.759	1.958
3.20	10.240	32.48	3726	27.48	0.440	5.47	17.50	0.818	21.38	1.757	1.958
3.30	10.890	32.16	3698	29.45	0.423	5.30	17.48	0.822	21.26	1.753	1.958
3.50	12.250	31.48	3639	33.66	0.391	4.97	17.41	0.830	21.00	1.747	1.953
3.75	14.062	30.57	3560	39.50	0.354	4.60	17.24	0.845	21.16	1.727	1.938

Properties of propellant:

$\mu = 23$
$\gamma = 1.23$
$T = 8.7 \times 10^{-5} h + 900$
$h_0 = 3.76 \times 10^7$ ft²/sec² = 1500 Btu/lb m
$R = 4.97 \times 10^4$
$n = 0.551$
$k = 5.5 \times 10^{-5}$ sec/ft for $f_1(\theta)$
$k = 6.5 \times 10^{-7}$ sec/ft for $f_2(\theta)$

amounts to matching the values of m_d and p_{sn} in Equation [8] with the values of m and p_s for the downstream end of the charge ($x = l$) from Equations [1]–[5] and [9]. The charge geometry is then introduced through Equation [7] and the relation thereby introduced between x and m.

Method of Solution

Solution of the Channel Flow Problem

In order to solve the Equations [1]–[5], first eliminate ρ between Equations [1] and [3] and solve for p/p_0. Then solve Equation [1] for ρ/p_0 and Equation [2] for m/p_0A_p.

$$p/p_0 = [1 + \mu u^2/RT]^{-1} \quad [10]$$

$$\rho/p_0 = p\mu/p_0RT \quad [11]$$

$$m/p_0A_p = \rho u/p_0 \quad [12]$$

Now if values of u are chosen, the following sequence of computations yields the pertinent variables: compute h from Equation [4], T from Equation [5], p/p_0 from Equation [10], ρ/p_0 from Equation [11], m/p_0A_p from Equation [12]. It is convenient to compute p_s/p_0 also at this point, using Equation [9] and the value of p/p_0 from the foregoing sequence.

Table I shows a sample computation following the above sequence with $\mu = 23$, $R = 4.97 \times 10^4$, $h_0 = 1500$ Btu/lb mass $= 3.76 \times 10^7$ ft²/sec², $\gamma = 1.23$, and $T = 8.7 \times 10^{-5} h + 900°$ R corresponding to Equation [5]. These values are typical of a propellant gas which does not dissociate appreciably at the temperatures and pressures involved. (Relations between enthalpy and temperature are obtained by solution of the equations of chemical equilibrium for composition of the propellant gas, and computation of the heat capacity by summing over the heat capacities of the component gases; γ is computed using the relations $C_p = dh/dT$, $C_v = R - C_p$, $\gamma = C_p/C_v$.) The data of Table I are shown graphically in Fig. 2. Because the quantity m/p_0A_p (denoted hereafter as θ) arises in subsequent manipulations of Equations [7] and [8], it is used as the independent variable. It is interesting to note that as θ is increased, the pressure decreases and the velocity increases up to a certain value of θ, and that further increases in velocity and decreases in pressure can be accomplished only by decreases in θ. This latter region corresponds to removal of mass from the stream, and is not of interest in the present problem where the only mechanism of mass addition is burning of the solid propellant. The critical value of θ corresponds to a value of u equal to sonic speed, as can be seen by substituting the value of T into $a \equiv (\gamma RT/\mu)^{1/2}$ and comparing a with u for the maximum value of θ (for ordinary rocket motors the value of θ is usually limited by ballistic considerations to values appreciably less than the

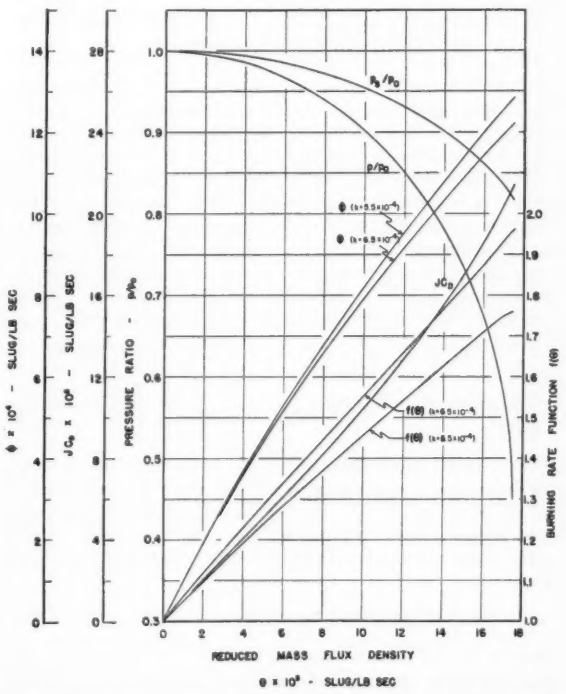

FIG. 2 ONE-DIMENSIONAL FLOW FIELD IN A SOLID PROPELLANT ROCKET MOTOR

critical value). It is of particular interest that the present method of solution of the flow problem does not require repetition of the solution for each value of upstream pressure p_0 and does not involve the burning characteristics of the propellant—as were the cases in (7). The virtue of this simplification will become increasingly apparent with increased familiarity with the computing processes and interpretation of results. As an example, it may be noted that the curve of p_s/p_0 vs. θ in Fig. 2 is analogous to one of the families of curves in Fig. 5 of (7), and this curve is not dependent on burning characteristics as are the members of the family of (7).

Solution of the Burning Problem

Since, for steady flow, the mass flux m in the channel at the point x must equal the net burning rate m_b upstream of the point x, $m = m_b$, and θ may be written

$$\theta = \frac{\rho_p q C}{p_0^{1-n} A_p} \int_0^x \left(\frac{p}{p_0}\right)^n (1 + ku)dx \quad \ldots\ldots [13]$$

using Equations [6] and [7] and dividing Equation [7] by $p_0 A_p$. This equation relates the independent variable θ of Fig. 2 to the geometrical variable x, so that the solution of Table I becomes a distribution of pressure, velocity, mass flux, temperature, and density as functions of x along the flow channel. The quantities ρ_p, q, C, A_p, n, and k are characteristics of the propellant and charge geometry.

To solve Equation [13], rewrite it

$$d\theta/d\varphi = f(\theta) = \left(\frac{p}{p_0}\right)^n (1 + ku) \quad \ldots\ldots [14]$$

where

$$d\varphi = (\rho_p q C/p_0^{1-n} A_p)dx$$

$$\varphi = (\rho_p q C/p_0^{1-n} A_p)x = \beta\, p_0{}^{n-1} \quad \ldots\ldots [15]$$

The dependence of $f(\theta)$ on θ is computed in columns 11 and 12 of Table I for values of k of 5.5×10^{-4} and 6.5×10^{-4} sec/ft, and is plotted in Fig. 2. These curves are used in solving Equation [14]. The variable φ in the present paper differs from the variable β in (1) by the factor $p_0{}^{n-1}$.

The method of Moulton (12) was used to solve Equation [14], and the resulting relations between φ and θ are shown in Fig. 2, using a value of n of 0.55 and values of k of 5.5×10^{-4} sec/ft and 6.5×10^{-4} sec/ft. It should be noted that the method of solution is applicable for any form of burning-rate law involving the stream variables of Equations [1]–[5]. The relation of φ to θ corresponds to a relation between x and the variables of state, since φ is proportional to x (up to this point, φ may be regarded either as position x on a charge, or as length of the charge, as the problem demands). It remains to apply the nozzle discharge formula [8] to determine what value of p_0 pertains in the proportionality factor relating φ to x in Equation [15].

Equilibrium Between Burning Rate and Nozzle Discharge Rate

For steady-state operation the nozzle discharge rate m_d equals m_b, where m_b is the value of m at the downstream end of the charge where $x = l$. Now in Equation [8] the stagnation pressure p_{sn} is closely approximated by the stagnation pressure p_s at the downstream end of the charge, and Equation [8] can be rewritten

$$(m_d/p_0 A_p)/(p_{sn}/p_0) = JC_D = \theta/(p_s/p_0) \quad \ldots\ldots [16]$$

where $J = A_t/A_p$ and Equation [8] was divided through by $p_0 A_p$ and by p_{sn}/p_0, and the above conditions applied. Since (p_s/p_0) is a known function of θ, Equation [16] gives JC_D as a function of θ. This relation is shown in column 10 of Table I and as a curve in Fig. 2. (In the event that $J > 1$, the flow at the nozzle throat is subsonic and flow at the downstream end of the charge is sonic, with θ equal to the maximum value on the θ vs. p/p_0 curve. Hence, JC_D should have the value C_D whenever $J > 1$.) It will be shown in the next section that combined results to this point enable one to compute p_0 and relate the φ scale to the position x on the charge axis, thus determining the flow field in the motor.

Combined Solutions

By combining the results of the three preceding sections, the flow field in the rocket motor can be specified provided the charge and motor geometry are established. The charge geometry is ordinarily known with good accuracy at the time reaction is initiated, and subsequent geometry can be computed using the solutions of the flow problem, so that the flow field can be determined as a function of time. The procedure is indicated in the following paragraphs.

The value of J can be computed from the motor-charge geometry. The value of C_D can be computed from the values of θ and p_s/p_0 for critical flow, since the conditions here are the same as those in the throat of the nozzle. In computing C_D, Equation [8] is modified to

$$C_D = \theta_{\max}/(p_s/p_0)_{\max} \quad \ldots\ldots [17]$$

by division by $p_{sn}A_p$, application of the definition $\theta = m/p_0 A_p$, and by setting $J = 1$ (this condition is based on the assumption of one-dimensional isentropic flow between the downstream end of the charge and the nozzle throat). For the solution shown,[3] $C_D = 2.19 \times 10^{-4}$ slug/lb sec. Using the computed values of J and C_D to obtain JC_D, θ for the downstream end of the charge can be determined from the JC_D vs. θ curve of Fig. 2, and the corresponding value of φ can be determined from the φ vs. θ curve. Then the value of the front pressure p_0 can be computed from Equation [15] in the form

$$p_0 = (\rho_p q C x/A_p \varphi)^{1/(1-n)} = (\beta/\varphi)^{1/(1-n)} \quad \ldots\ldots [18]$$

where the value of x is that corresponding to the downstream end of the charge (i.e., $x = l$). The quantity β contains the factor ql/A_p, descriptive of the geometry of the propellant charge, which is frequently referred to as the "internal area ratio."

Using a value of C_D of 2.19×10^{-4} slug/lb sec, Fig. 2 was used to find φ for several values of J; for each value of J the corresponding value of φ was used in Equation [18] to determine the dependence of p_0 on β, and the results are shown in Fig. 3. If the factor 12^{2-2n} in the abscissa is omitted, the pressure scale will correspond to lb/ft². For values of $J > 1$, the curve for $J = 1$ is applicable, since sonic flow prevails at the exit for $J \geq 1$, and the nozzle exerts no influence on the flow field upstream of the sonic surface.

The foregoing computation of p_0 enables one to (a) establish a correspondence between φ and x for the rest of the flow field by Equation [15], and (b) eliminate the p_0 in the ratios p/p_0, p_s/p_0, ρ/p_0, and $m/p_0 A_p$ in Table I and Fig. 2, so that the actual flow field is determined. Furthermore, it enables one to compute the burning rate of Equation [6], since, using Equation [14]

$$r = Cp^n(1 + ku) = Cf(\theta)p_0^n \quad \ldots\ldots [19]$$

and $f(\theta)$ is obtainable from the $f(\theta)$ vs. θ curve.

Since the burning rate of Equation [19] is not the same everywhere along the flow channel, a charge which initially has the same cross section over its entire length will not remain exactly so shaped during burning, and the theory of this paper will be somewhat less accurate. However, the deviation from assumed geometry is not great with most designs, and good

[3] This procedure for computing C_D is equivalent to use of Equation [23], p. 40, of (1).

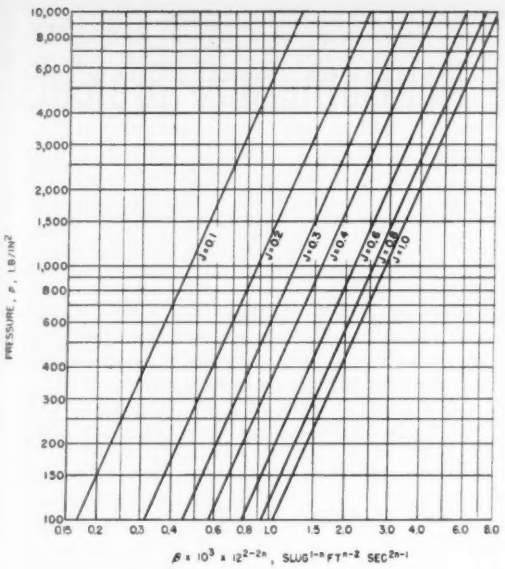

FIG. 3 DEPENDENCE OF UPSTREAM PRESSURE ON THE PARAMETERS β AND J

agreement between computed and observed pressure-time curves has been obtained by computing the charge geometry on the assumption that the burning rate has the same value everywhere on the charge, the value being that given by Equation [19] with θ corresponding to the downstream end of the charge. This assumption has a twofold purpose. First, it approximates the actual geometry by that pertinent to the present theory. Second, it greatly simplifies the computation of the charge geometry at the various times during burning. The adequacy of the assumption should be examined for designs resulting in large values of θ.

Comments and Discussion

The question of adequacy of initial assumptions naturally must be examined when the methods described here are applied to a particular problem. No detailed discussion is intended here, but it should be noted that the effects of viscosity will cause appreciable error when high stream velocities are involved. The effects of nonuniformity of velocity in planes normal to the x-axis will also affect the accuracy of the calculations. These effects cannot be evaluated accurately because the details of the processes are not known in advance. Nonetheless, the results presented here are more accurate than the conventional procedure in rocket work of simply neglecting pressure variations in the combustion chamber, and the errors due to the above causes are comparable with other sources of error (e.g., in knowledge of burning rate of the propellant). For values of J greater than 0.7 where the above sources of error become appreciable, the analysis yields a qualitative description of behavior of rocket motors which has proven most useful in understanding motor performance. Even for $J \geq 1$, these sources of error exert only secondary effects.[4]

It is also worthy of note that the general procedures outlined in this paper are applicable for other mass addition rate laws, the only change in procedure being a suitable redefinition of φ and β to include whatever factors can be removed from the integrand in Equation [13]. The function $f(\theta)$ is determined by the dependence of the mass addition rate on the stream variables, and if the restricted form of the burning rate rule in Equation [6] is used, $f(\theta)$ is the ratio of the burning rate r at the point x to the burning rate r_0 at the point $x = 0$.

[4] However, in rocket work these large values of J have not been used because of excessive mechanical deformation of the propellant charge due to large pressure differences in the combustion chamber.

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Maximum Evaporation Rates for Nonisothermal Droplets

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Calculations have been carried out in order to determine the rate of evaporation of a liquid droplet surrounded by hot gases. The present study represents an extension of earlier work by Penner on evaporation rates for isothermal droplets. In particular, allowance was made for temperature gradients within the droplet by considering a droplet composed of an isothermal core and an isothermal shell. The results obtained in the present investigation were found to be in satisfactory agreement with the known data for evaporation of isothermal droplets, thus justifying calculations for isothermal droplets as a reasonable first approximation.

Introduction

THE purpose of this study is to extend earlier work on evaporation rates of isothermal droplets in rocket engines. Penner (1)[2] treated the evaporation rate of a liquid droplet in a gas at constant temperature assuming Stokes flow. The limitations involved in this approximation, as well as references to the original literature, have been stated in (1) and need not be repeated here. Although the droplet temperature changed with time, it was assumed that the droplet remained isothermal during evaporation, i.e., that the thermal conduction coefficients of evaporating liquid droplets were, for all practical purposes, infinite.

We have considered a shell model of the evaporating droplet in which the droplet is divided into two parts, an inner core and an outer spherical shell. The inner sphere was assumed to be isothermal and to remain at the original temperature of the evaporating liquid; the outer shell was also assumed to be isothermal, but its temperature was determined by making an appropriate heat balance equation. The thickness of the spherical shell was treated as a variable parameter, and it was found that results substantially equivalent to those obtained for isothermal droplets are derived, independently of the thickness of the spherical shell.

Of the evaporating droplets treated previously (1) by assuming isothermal conditions during evaporation, we have chosen for study an aniline droplet with initial radius equal to 5×10^{-3} cm. The initial temperature was set equal to 300° K. Radiant heat transfer to the droplet was neglected since it is probably unimportant in liquid-fuel rocket motors (1).

The Shell Model of the Evaporating Droplet[3]

The concept of an isothermal droplet during evaporation in combustion chambers represents a first approximation. In practice one would expect that temperature gradients are set up during the life of the droplet. If the thermal conductivity is sufficiently small, then it is clear that steep temperature gradients may be set up near the surface of the droplet. However, the surface temperature is not necessarily higher than for an isothermal droplet because it is determined by a heat balance involving cooling by evaporation.

As a first approximation to a droplet with temperature gradients, it was assumed that an evaporating droplet could be represented by an isothermal core surrounded by an isothermal shell. The shell thickness ϵ (cm) was set equal to 0.25, 10^{-2}, and 10^{-5} times the original droplet radius and was assumed to remain constant during evaporation for any given droplet.

In general the surface temperature will be determined by an energy balance between heat input into the droplet and absorption of energy by evaporation or by thermal conduction to the core of the spherical droplet. The rate of heat transfer Q (cal/sec) to a sphere of radius r (cm) is

$$Q = 4\pi r^2 h \Delta T \quad \ldots\ldots [1]$$

where ΔT(°K) represents the temperature difference between the gases surrounding the liquid droplet and the surface temperature of the liquid droplet, and h (cal/cm² sec °K) is the over-all coefficient of heat transfer to a liquid droplet evaluated at the mean film temperature $T_f = T_1 + {}^1/_2(T_g - T_1)$. Here T_g (°K) is the temperature of the hot gases, and T_1(°K) is the temperature of the surface of the liquid.

For the simplified evaporation model employed for the present studies, all of the energy input occurs into a shell of volume

$$V = 4/3\pi r^3 - 4\pi \int_0^{r-\epsilon} r^2\, dr = (4\pi/3)r^3[1 - (1 - \epsilon/r)^3] \quad \ldots [2]$$

From Equation [2] it is seen that for $\epsilon/r \ll 1$, $V \simeq 4\pi r^2 \epsilon$; for $\epsilon/r = 1$ (isothermal droplet), $V = (4\pi/3)r^3$.

It is apparent from Equations [1] and [2] that if heat transport to the spherical droplet were the only important physical process, then the temperature T_1 in the isothermal shell of thickness ϵ would rise at a rate

$$(dT/dt)_c = 3h\Delta T/r[1 - (1 - \epsilon/r)^3]\rho c \quad \ldots\ldots [3]$$

where ρ (gm/cc) and c (cal/gm) represent, respectively, the density and heat capacity of the liquid in the isothermal shell. For $\epsilon/r \ll 1$, Equation [3] becomes $(dT/dt)_c = h\Delta T/\epsilon\rho c$; for $\epsilon/r = 1$ (isothermal droplet), $(dT/dt)_c = 3h\Delta T/r\rho c$.

We assumed that ϵ remains constant as the droplet radius r decreases during evaporation. The rate of decrease of volume by evaporation of the isothermal shell of thickness ϵ is

$$-dV/dt = 4\pi r^2(-dr/dt) \quad \ldots\ldots [4]$$

The rate of absorption of heat by evaporation Q_e (cal/sec) is evidently

$$Q_e = (-dV/dt)(\rho l) \quad \ldots\ldots [5]$$

where l (cal/gm) equals the heat of evaporation of the liquid in the shell. Since the total heat capacity of the shell is $V\rho c$, the rate of decrease of temperature in the shell, if only evaporation occurred, is

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[1] Captain, USAF. This paper is abstracted from a thesis submitted by F. W. Hartwig to the graduate school of the California Institute of Technology in partial fulfillment of requirements for the degree of Aeronautical Engineer, June 1952. Present address: U. S. Air Force Institute of Technology, Wright-Patterson Air Force Base, Dayton, Ohio.

[2] Number in parentheses refers to the Reference on page 243.

[3] The author is indebted to Dr. S. S. Penner for help with the formulation of the shell model of the evaporating droplet.

$$-(dT/dt)_e = (3l/cr)(-dr/dt)[1 - (1 - \epsilon/r)^3]^{-1} \ldots [6]$$

The model of the evaporating droplet adopted in the present discussion requires allowance for another heat sink. Thus the assumption that ϵ is constant means that the inner surface of the isothermal shell must travel in the direction of the center of the sphere sufficiently rapidly to maintain ϵ constant. This travel of the inner surface of the outer shell means that some mass will be introduced from the colder core of the droplet into the outer shell. Since the shell must remain isothermal, heat must be added to this new mass to bring it up to the temperature of the shell. The surface area $4\pi(r - \epsilon)^2$ moves inward with a velocity $(-dr/dt)$. Hence the energy absorbed per unit time in order to keep ϵ constant and to keep the shell isothermal is

$$4\pi(r - \epsilon)^2(-dr/dt)\rho c(T_1 - T_0)$$

In order to correct for this heat sink, we may say that of the total heat Q transferred to the droplet in unit time, only the amount

$$Q' = 4\pi r^2 h\Delta T - 4\pi(r - \epsilon)^2(-dr/dt)\rho c(T_1 - T_0) \ldots [1a]$$

is effective in producing evaporation and heating the outer isothermal shell.

If Equation [1a] is used in place of Equation [1], Equation [3] becomes

$$(dT/dt)_c = 3h\Delta T/r[1 - (1 - \epsilon/r)^3]\rho c - 3(1 - \epsilon/r)^2(-dr/dt)\,(T_1 - T_0)/r\,[1 - (1 - \epsilon/r)^3] \ldots [3a]$$

Neglecting heat transfer between the isothermal regions, it is apparent that

$$(dT/dt) = (dT/dt)_c + (dT/dt)_e$$

represents the rate of change of temperature in the isothermal shell with time. Using Equations [3a] and [6] we obtain the result

$$dT/dt = \{3/r[1 - (1 - \epsilon/r)^3]\}\{(h\Delta T/c\rho) - (1 - \epsilon/r)^2(-dr/dt)(T_1 - T_0) - (l/c)(-dr/dt)\} \ldots [7]$$

Similarly, for $\epsilon/r \ll 1$

$$dT/dt = (h\Delta T/\epsilon\rho c) - (-dr/dt)[(l/c\epsilon) + (T_1 - T_0)/\epsilon] \ldots [7a]$$

and for $\epsilon/r = 1$

$$dT/dt = 3h\Delta T/r\rho c - (3l/cr)(-dr/dt) \ldots\ldots [7b]$$

Equation [7b] represents the known result for isothermal droplets (1).

In order to carry out approximate calculations for the rate of change of temperature T_1 and droplet radius r with time, we may assume that $-dr/dt$ is given by the Knudsen equation with an evaporation coefficient α set equal to unity. Thus

$$-dr/dt = (p/\rho)(M/2\pi RT)^{1/2} \ldots\ldots\ldots\ldots [8]$$

where p is the saturated vapor pressure of the liquid at the temperature T_1, M is the molecular weight, and R is the molar gas constant. The pair of Equations [7], [7a], or [7b] together with Equation [8] can be solved by a simple iterative procedure (1).

Representative numerical values for aniline are the following: $c = 0.53$ cal/gm °K; $r° = 5 \times 10^{-3}$ cm; $h = k/r = 4.30 \times 10^{-4}/r$ cal/cm² sec °K; $\Delta T = (3000 - T_1)$ °K; $\rho = 1.03$ gm/cc; $l = 129$ cal/gm; $-dr/dt = \text{antilog}_{10}\,(7.09 - 2600/T_1)$; $T_0 = 300$ °K; $T_f = 1650$ °K. The results of the present calculations for $\epsilon/r = 0.25$, 10^{-2}, and 10^{-4} are plotted in Fig. 1 together with the known results for $\epsilon/r = 1$ (isothermal droplet).

Reference to Fig. 1 shows that the smaller the value of ϵ/r, the faster the shell temperature increases. However, the higher the temperature, the faster the evaporation rate.

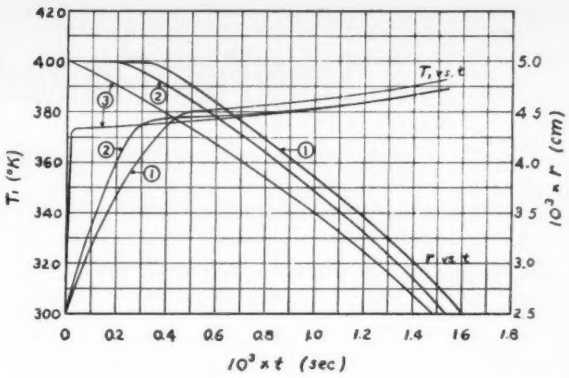

FIG. 1 DROPLET RADIUS AND SURFACE TEMPERATURE AS FUNCTIONS OF TIME FOR AN EVAPORATING DROPLET

① $\epsilon/r = 1$ ② $\epsilon/r = 1/4$ ③ $\epsilon/r = 10^{-2}$ or 10^{-4}

Therefore, although the temperature rises faster initially in the thinner shells, a pseudo steady-state temperature is apparently reached at a slightly lower temperature than for an isothermally evaporating droplet. In general, reference to Fig. 1 shows that the results obtained from the shell model are consistent with those obtained for the isothermal case. Hence it may be justified to conclude that calculations of evaporation rates, based on the isothermal approximation, are reasonable.

A still more realistic approximation to an evaporating droplet has been given by H. S. Tsein who has represented the temperature profile within an evaporating droplet by a parabolic distribution. The solution of the resulting set of equations is quite laborious;[4] the formulation can be improved by utilizing a temperature profile corresponding to very large temperature gradients near the droplet surface.

[4] For details concerning this work, see the AE thesis of F. W. Hartwig, California Institute of Technology, June 1952.

Reference

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A Simplified Combustion Analysis System

R. K. NEUMANN,[2] D. DEMBROW,[3] W. G. BERL,[3] and R. PRESCOTT[3]

Applied Physics Laboratory, The Johns Hopkins University, Silver Spring, Md.

Knowledge of the chemical composition of the inlet and exhaust gases of combustion engines aids in evaluating engine performance. Jet engine research requires rapid gas analysis unobtainable by conventional analytical methods. A convenient and rapid gas analysis system which combines thermal conductivity measurements with a chemical analytic scheme and which has been employed extensively on ramjet engines is described. Gases are analyzed for fuel-oxidizer ratio and for chemical composition. With simplifying assumptions, this analysis permits the calculation of other performance data such as combustion efficiency, flame temperature, Mach number, etc.

Introduction

IN THE development of jet engines, over-all performance measurements such as total thrust are frequently not informative enough to give insight into the detailed operation of such engines. A knowledge of local fuel-air ratios and combustion conditions within the engine is of considerable aid in unraveling the complex situations encountered. This paper describes an analytical scheme and apparatus which have proved useful in the determination of the local composition of the working fluid in a jet engine. With this information it is possible to compute local combustion efficiencies and temperatures. With associated pressure measurements, flow parameters such as local Mach numbers and velocities can be determined.

The principle employed in this instrument is that of measuring the concentration of a stable component of the combustion reaction after suitable chemical operations have been performed on the sample. Table 1 supplies some comparative characteristics of a variety of analytical techniques which could be used for determining the concentration of this stable component. The evaluation is based on the analysis for carbon dioxide in a hydrocarbon-air system. Different fuel-oxidizer combinations may require revisions in this evaluation.

TABLE 1

Technique	Simplicity	Cost	Rapidity	Versatility
Mass spectrometer	−−	−−	±	++
Infrared	+	±	+	−
Orsat	++	++	−−	+
Paramagnetic oxygen analyzer	+	+	+	−
Thermal conductivity	+	+	+	+

On the basis of the comparison in Table 1, the thermal conductivity technique was chosen as most suitable, and carbon dioxide selected as the stable component in view of the large variation of its thermal conductivity with concentration (1).[4] The subsequent discussion will be based on an instrument designed for hydrocarbon-air combustion analysis. Suitable modifications will make it applicable to other chemical systems.[5]

Description of Method

The following components are likely to be present during the combustion of a hydrocarbon fuel: carbon dioxide, carbon monoxide, unreacted fuel,[6] water, nitrogen,[7] hydrogen, and oxygen. Such a mixture cannot be analyzed directly since thermal conductivity fixes composition uniquely only in a two-component system. If more than two components are present, other relations and conditions must be established to make unique solutions possible. For this reason a chemical analytical scheme is introduced to transform the original sample into a three-component system consisting solely of carbon dioxide, nitrogen, and oxygen, and to obtain additional relations by appropriate chemical means. As shown

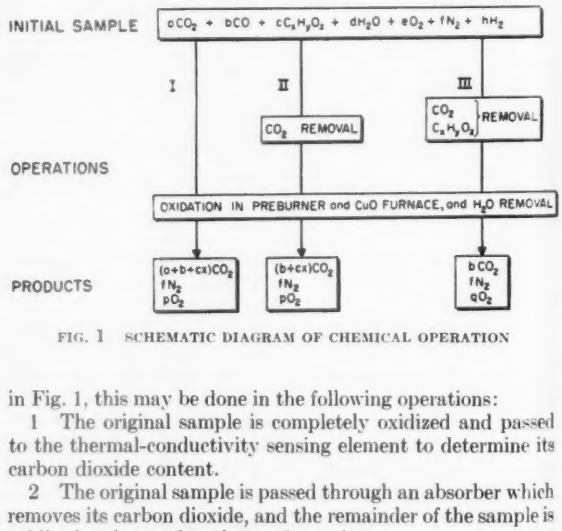

FIG. 1 SCHEMATIC DIAGRAM OF CHEMICAL OPERATION

in Fig. 1, this may be done in the following operations:

1 The original sample is completely oxidized and passed to the thermal-conductivity sensing element to determine its carbon dioxide content.

2 The original sample is passed through an absorber which removes its carbon dioxide, and the remainder of the sample is oxidized and passed to the sensing unit.

3 The original sample is passed through an additional absorber which removes any hydrocarbons (unreacted fuel), and the remainder of the sample is oxidized and passed to the sensing element.

In all cases, water vapor is removed from the sample by a dehydrating agent before reaching the thermal conductivity cell. Any oxygen deficiency in the original sample is compensated for by passing the sample over hot copper oxide wire where sufficient oxygen for complete oxidation is supplied without introducing any free oxygen into the system.

Presented at the 7th Annual Convention of the AMERICAN ROCKET SOCIETY, New York, N. Y., December 3, 1952.

[1] This work is supported under contract NOrd 7386 with the Bureau of Ordnance, U. S. Navy.

[2] Research Mathematician. Member ARS.

[3] Research Chemist.

[4] Numbers in parentheses refer to the Bibliography on page 247.

[5] See Bibliography for a list of references to other gas-analysis systems using thermal conductivity techniques (2–6).

[6] Intermediates such as methane and aldehydes are included in this group.

[7] Includes other inert gases such as argon, krypton, etc.

Description of Apparatus

Sensing Element. The sensing element is a thermal conductivity cell[8] consisting of four heated wires connected as a Wheatstone bridge. Two of the wires are immersed in dry air, the other two are contained in recessed chambers into which the unknown gas mixture is allowed to diffuse. If all wires are surrounded by air, the bridge can be balanced at some convenient wire and cell temperature. Whenever a sample of different thermal conductivity surrounds two of the wires, their temperature and resistance will change and unbalance the bridge. This unbalance is indicated by a galvanometer. The magnitude of this unbalance is a measure of the thermal conductivity of the unknown sample relative to that of the standard gas (air), and is also a function of the composition of the gas sample.

In view of the reproducibility and stability required of the instrument, special care must be taken to reduce the causes of zero instability and variations in the response of the bridge circuit. In order to achieve adequate accuracy it is necessary that the bridge current (or voltage) remain constant and that the thermal conductivity unit be placed in a thermostatically controlled cabinet.

The response time of such a bridge has been investigated by Daynes (1) and found to be exponentially dependent upon the rate of diffusion of the gases to the cell containing the thermal elements. By reducing the diffusion path to a minimum, the response time is short and 99 per cent of full-scale deflection can be reached in about 30 seconds.

Chemical Analysis

The analytical train consists of two furnaces and several chemical absorbents (Fig. 2). The function of the first furnace is to complete the oxidation of any unoxidized or partially oxidized materials with any available oxygen which may be present in the original sample. This is done in a platinum-iridium tube coiled into a helix and maintained at approximately 1600° C by resistance heating. The oxidation is completed in a second furnace which supplies any additional oxygen necessary for complete oxidation, and consists of copper oxide wire in a tube maintained at 700° C. Two such copper oxide furnaces are necessary, one being regenerated with air while the other is used for analysis. Beyond an equivalence ratio[9] of 3 for most hydrocarbons, it is necessary

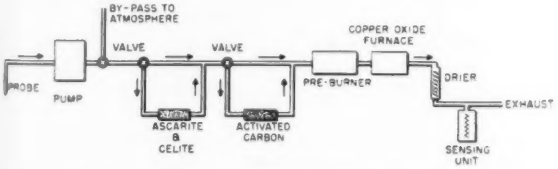

FIG. 2 SCHEMATIC DIAGRAM OF ANALYTIC SCHEME

to dilute the sample with air ahead of the first furnace to prevent carbon deposition there.

For convenience in handling, solid rather than liquid absorbents are used in the system. These absorbents are packed into glass tubes and the gas allowed to filter through them. Water is removed ahead of the thermal conductivity cell by means of granular magnesium perchlorate. Carbon dioxide in the initial sample is removed by Ascarite (a commercially available solid sodium hydroxide-asbestos mixture) diluted with Celite. Hydrocarbons are adsorbed by passage through activated carbon. These agents become saturated with use and must be periodically replaced.

[8] Supplied by Gow-Mac Instrument Co., Newark, N. J.

[9] Equivalence ratio ϕ is defined as the actual fuel/air ratio divided by the stoichiometric fuel/air ratio.

Associated Equipment. The complete instrument requires the following accessories: A pump is necessary to push the gas sample through the chemical absorption train and the thermal conductivity meter. The pump and connecting fittings must contain no materials (rubber, oil, etc.) which may act as absorbents or contaminants, and must be leakproof. Paint spray pumps, equipped with neoprene diaphragms, are satisfactory for most hydrocarbons.

Sample lines must be maintained above the temperature at which any of the components in the sample might condense (except water) and yet low enough to prevent further reaction. The solubility of carbon dioxide in any condensed water is negligible.

Transfer valves permit routing of the sample through the various absorbents and furnaces and allow alternate regeneration of the exhausted copper oxide furnaces. Flowmeters are desirable for maintaining optimum flow rates for minimum response time.

Calibration and Data Reduction

In the conversion of the thermal conductivity measurements into chemical composition the following assumptions are made: (a) The concentration of free radicals and dissociation products in the original sample is small enough to be neglected. (b) The entire carbon content of the sample is convertible to carbon dioxide. This implies no losses due to condensation, undesired absorption, carbon precipitation, etc. (c) The compositions of fuel and oxidizer are known. No elements other than carbon, hydrogen, oxygen, and nitrogen are present.

The following nomenclature will be used. The fuel composition is $C_xH_yO_z$. The oxidation reaction is

$$C_xH_yO_z + gO_2 + fN_2 \rightarrow aCO_2 + bCO + cC_xH_yO_z + dH_2O + eO_2 + fN_2 + hH_2$$

The fraction of the carbon completely burned to CO_2 is (a/x); the fraction partially oxidized to CO is (b/x), and the fraction unreacted is (c). Let $(x + y/4 - z/2) = K$. If all the carbon is oxidized to CO_2, the following equations apply

Lean case, $\phi < 1$ (excess air):

$$\frac{CO_2}{CO_2 + O_2 + N_2} = \frac{x}{x - K + 4.764\,\frac{K}{\phi}} \quad \ldots\ldots [1]$$

Rich case, $\phi > 1$ (excess fuel):

$$\frac{CO_2}{CO_2 + N_2} = \frac{x}{x + 3.764\,\frac{K}{\phi}} \quad \ldots\ldots [2]$$

Calibration is carried out by running several samples of different fuel-rich mixtures through operation I (Fig. 1), determining the meter deflection for each, and making an accurate chemical analysis of the carbon dioxide content of the oxidized mixture by some analyzer such as an Orsat unit. The fraction CO_2/total gas is then plotted against the corresponding meter reading. This establishes the slope N of line $\overline{AP}$ and its intercept θ_T (Fig. 3).

The carbon dioxide concentration, corresponding to the completely oxidized stoichiometric fuel/air mixture (σ_0), is calculated from either Equation [1] or [2], $(\phi = 1)$. This determines point A and the corresponding meter reading θ_0. The slope M of line $\overline{OA}$ is determined. The meter deflection due to atmospheric nitrogen θ_N is obtained by passing fuel-rich mixtures through operation II (Fig. 1) and removing all carbon dioxide and water ahead of the sensing unit.

Fuels of differing H/C and O/C ratios require separate calibrations. Care must be taken to insure that enough experimental points are established to give an accurate and meaningful calibration.

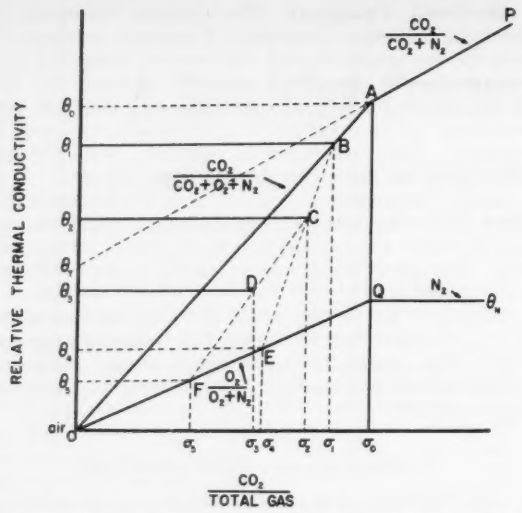

FIG. 3 RELATIONSHIP BETWEEN CARBON DIOXIDE CONTENT AND THERMAL CONDUCTIVITY FOR A FUEL-AIR SYSTEM

The equivalence ratio ϕ can be determined from the meter reading θ_1 of operation I

Lean case ($\theta_1 < \theta_0$):

$$\phi = \frac{\theta_1}{M}\left(\frac{4.764\,K}{x + \frac{\theta_1}{M}(K - x)}\right) \quad \ldots\ldots [3]$$

Rich case ($\theta_1 > \theta_0$):

$$\phi = \frac{3.764\,K(\theta_1 - \theta_T)}{x(N + \theta_T - \theta_1)} \quad \ldots\ldots [4]$$

The fraction of carbon as CO_2 in the original mixture (a/x) is a function of the difference of the meter readings ($\theta_1 - \theta_2$) of operations I and II

$$\frac{a}{x} = \frac{\theta_1 - \theta_2}{\theta_1 - \theta_4} \quad \ldots\ldots [5]$$

where

$$\theta_4 = \theta_N\left(1 - \frac{1 - \phi}{1 - 0.21\,\phi}\right)$$

The fraction of carbon as unburned fuel (c) is a function of the difference of the readings $\theta_2 - \theta_3$ of operations II and III

$$c = \frac{\theta_2 - \theta_3}{\theta_2 - \theta_5}\left(1 - \frac{a}{x}\right) = \frac{\theta_2 - \theta_3}{\theta_2 - \theta_5} \times \frac{\theta_2 - \theta_4}{\theta_1 - \theta_4} \quad \ldots\ldots [6]$$

where

$$\theta_5 = \theta_N\left(1 - \frac{1 - \phi(a/x)}{1 - 0.21\,\phi(a/x)}\right)$$

The fraction of carbon as CO, (b/x), is found by difference

$$\frac{b}{x} = 1 - \frac{a}{x} - c \quad \ldots\ldots [7]$$

or, independently

$$\frac{b}{x} = \frac{\theta_3 - \theta_5}{\theta_2 - \theta_5}\left(1 - \frac{a}{x}\right) \quad \ldots\ldots [8]$$

Equations [5], [6], and [8] are true for the lean case only. The fuel-rich case is similar but differs in the constants applied.

From the chemical mass balance equations the composition of the remainder of the gas may be determined if it is assumed that hydrogen and oxygen do not coexist in the products and will combine to form water until one or the other component is exhausted.

Knowing the composition, an adiabatic flame temperature can be computed by evaluating the formula

$$\left[\sum_i (h_i N_i)_{\text{reactants}}\right]_{T_0} = \left[\sum_i (h_i N_i)_{\text{products}}\right]_{T_f} \quad \ldots\ldots [9]$$

where h_i is the specific enthalpy which includes the standard heat of formation, and N_i is the quantity of the ith component.

The combustion efficiency (η_c) is defined as the ratio of the actual heat release to the heat release obtained if the reaction had gone to equilibrium and may be found as follows

$$\eta_c = \frac{\left[\sum_i (h_{f_i} N_i)_{\text{prod}} - \sum_i (h_{f_i} N_i)_{\text{reac}}\right]_{\text{actual}}}{\left[\sum_i (h_{f_i} N_i)_{\text{prod}} - \sum_i (h_{f_i} N_i)_{\text{reac}}\right]_{\text{ideal}}}$$

where h_{f_i} is the heat of formation of the ith component.

These methods of computing flame temperature and efficiency are somewhat long and tedious. If repeated computations are to be done, a selected group of mixtures may be calculated in detail and nomographs for intermediate mixtures can be prepared. Examples of this are shown in Figs. 4a, 4b. These plots are for a specified fuel, mixture ratio (ϕ), and initial temperature. By stepwise variation of these conditions, other plots are obtained from which interpolations can be made.

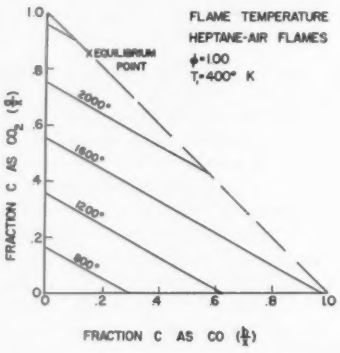

FIG. 4a

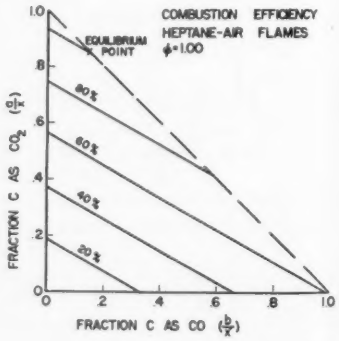

FIG. 4b

FIGS. 4a, 4b NOMOGRAPH OF FLAME TEMPERATURE AND COMBUSTION EFFICIENCEY AS A FUNCTION OF CARBON DIOXIDE AND CARBON MONOXIDE CONTENT OF THE SAMPLE

From a knowledge of gas composition the molecular weight, specific heat, etc., can be found.

Discussion of Results

As a check on the validity of the assumptions and on the accuracy of the system itself, various comparisons have been made with results obtained by other systems. Four differ-

ent combustion zone samples were analyzed by the meter and also by a mass spectrograph.[10] The mass spectrograph also analyzed for minor constituents and intermediates such as ethylene, acetylene, etc. Table 2 shows a comparison of results obtained by the two methods.

TABLE 2 COMPARISON OF THERMAL CONDUCTIVITY METER WITH MASS SPECTROGRAPH ANALYSES

Sample no.	Adiabatic flame temp, °R — Therm. cond.	Adiabatic flame temp, °R — Mass spectro.	Fraction C as CO_2 — Therm. cond.	Fraction C as CO_2 — Mass spectro.
1	3040	2970	0.537	0.529
2	3110	3130	0.537	0.564
3	4100	4070	0.842	0.855
4	2380	2400	0.483	0.466

Temperatures calculated from the meter have been compared with temperatures obtained by other devices such as shielded thermocouples and pneumatic probes (7), Fig. 5. The temperatures calculated on the basis of the chemical composition of the sample assume that the entire heat of reaction has been retained in the sample. Temperature changes due to cooling by heat transfer, radiation, or external work cannot, therefore, be detected.

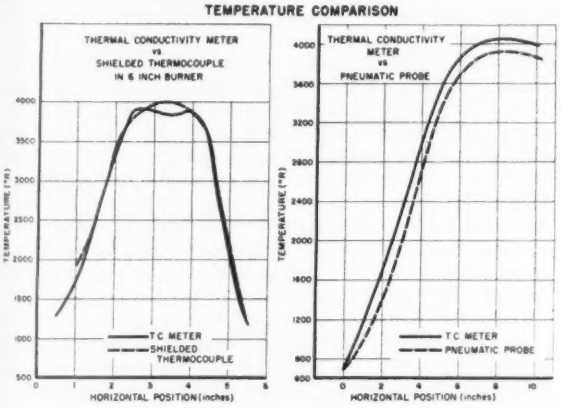

FIG. 5 COMPARISONS OF CALCULATED TEMPERATURES FROM CHEMICAL ANALYSIS WITH MEASURED TEMPERATURES

Fig. 6 shows a typical survey across the exit of a ramjet engine pointing up regions of combustion inefficiency and of nonuniformity in air-fuel distribution. Such analyses can be integrated to give the over-all performance. The comparison of such integrations with independently measured quantities such as air and fuel flows and thrust is also shown in Fig. 6.

Conclusions

Despite the simplifying assumptions, with proper care and maintenance the gas analysis instrument can measure air-fuel ratios in approximately 60 seconds with adequate accuracy. A complete chemical analysis (excluding data reduction) of the carbon-containing components can be made in three minutes.

Data reduction has been made simple and rapid with the aid of graphical solutions. Data from one analysis can be

[10] Analyses were carried out through the courtesy of Dr. M. Shepherd, National Bureau of Standards, Washington, D. C.

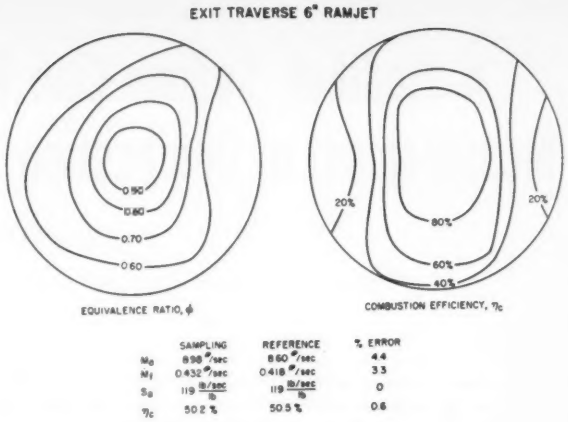

FIG. 6 EXIT SURVEY OF A TYPICAL RAMJET POWER PLANT

reduced to performance parameters including specific impulse, Mach number, velocity, and mass flows in 10 to 15 minutes. Comparisons with performance data obtained by temperature and thrust measuring devices have agreed within 5 per cent.

The gas analysis instrument described has been used extensively for ramjet testing and basic combustion research. It can be applied directly to other hydrocarbon-air combustion engines, and with simple alterations of the chemical analytic scheme and/or data reduction procedure can be used for other fuel-oxidizer systems.

The analyzer has been built into a rugged, self-contained, movable unit, well suited for laboratory testing. The operating procedure is simple and does not require highly trained personnel for operation. It can be used to provide discrete or continuous determination of the chemical composition of the working fluid of jet propulsion engines.

Acknowledgments

The help and assistance of Mr. C. Grunfelder and Mrs. J. Chalifoux are gratefully acknowledged.

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A Modified Sodium-Line Reversal Technique for the Measurement of Combustion Temperatures in Rocket Engines

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The application of a modified sodium-line reversal technique to the measurement of combustion temperature at several stations within an open tube rocket combustor was investigated. The measurement technique, experimental apparatus, and results are described and discussed. The effect of oxidant-fuel ratio on axial temperature distribution for a single two-on-one impingement type injector utilizing liquid oxygen and alcohol as propellants was evaluated. The experimental results indicate that the technique may be successfully applied to the study of combustion reaction time in rocket engines.

Introduction

THE development of a combustion chamber and propellant injector combination producing high efficiency with minimum chamber volume is one of the basic problems in the development of a rocket engine installation. The primary variables in such development are the propellant injection configuration and chamber geometry. The effect of changes in these parameters on combustion performance is only empirically defined for some propellant combinations and almost unknown for others. Extensive developmental programs are, therefore, required for each engine before optimum conditions are achieved. Such development is further complicated because over-all engine performance is generally the only measurement available for evaluating combustion efficiency. Each over-all performance measurement evaluates the effect of injector configuration for a particular combustion chamber configuration. Therefore, such effects as changes in nozzle efficiency mask the effects due to propellant injector configuration and chamber geometry. Because of the multiplicity of the variables involved in such evaluations, a more direct approach to the problem is desired. One method of attack that would provide basic information on the effect of changes in injection configuration is an evaluation of the completeness of combustion as a function of the axial chamber length. With such information the effect of changes in injection configuration on the combustion process, rather than on over-all performance, would be determined, and application of the data to other chamber lengths or to an increase in engine scale could be attempted with greater confidence.

An evaluation of the completeness of combustion at a particular station along the axial length of a combustion chamber requires a knowledge of the gas flow conditions at that station. One attempt to evaluate the gas flow conditions (1)[2] involved the evaluation of gas velocity by slit photography. It was shown that the axial velocity profile varied with injector configuration. Velocity may also be evaluated photographically by the technique described in (2). Velocity itself, however, does not describe the completeness of combustion without a knowledge of gas density and gas mass flow. Another indication of completeness of combustion is the gas temperature. Although gas temperature is admittedly an important parameter in the study of the combustion process, the measurement of temperature is made difficult due to the high temperature, velocity, and pressure conditions existing in the combustion chamber. Also, thermal equilibrium probably does not exist in the presence of combustion reactions. Nevertheless, a temperature measuring technique such as the sodium-line reversal method may indicate the extent to which reaction has occurred in a rocket chamber. It is the purpose of this investigation to study the utility of an optical technique for the measurement of temperature in a rocket combustor.

A modified sodium-line reversal technique was the optical system considered in this investigation. It is an adaptation of a technique described and analyzed in (3), and embodies many of the basic principles evolved from the standard sodium-line reversal method (4, 5). A similar method utilizing an OH spectral line rather than the sodium line was successfully applied to the exhaust of a rocket engine (6). Infrared radiation was used successfully by Silverman (7) for temperature measurements in a laboratory flame. Adaptation of such optical methods to gases within the rocket chamber, however, has not been reported.

In this investigation the utility of a modified sodium-line reversal technique was evaluated by applying the technique to the measurement of temperatures at various stations in an open tube rocket combustor with liquid oxygen and alcohol as propellants. Measurements were made covering a range of oxidant-fuel ratios. The measuring technique, experimental apparatus, and results are described and discussed.

Temperature Measurement Technique

Method of Temperature Measurement

The method used for the measurement of temperature in the rocket combustor is an optical method requiring that portion of the flame under investigation to be visible to the measuring apparatus. This method only utilizes the flame's radiation intensity in a finite spectral band. Flame temperatures are calculated from variations in radiation intensity produced by intermittently transmitting light from a tungsten lamp through the combustion flame. The degree of variation in radiation intensity is a function of the absorptivity and emissivity of the flame and the intensity of the tungsten light source. From known radiation theory the variations in radiation intensity are sufficient to calculate actual flame temperature. This method of temperature measurement is described in greater detail in (8). The necessary theory required for the application of the method is included in a succeeding portion of this paper.

Description of Instrumentation

The instrumentation used to produce and measure the variation in flame radiation intensity consists essentially of a

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Presented at the Annual Meeting of the AMERICAN ROCKET SOCIETY, New York, N. Y., December 5, 1952.

[1] Aeronautical Research Scientist. Member ARS.

[2] Numbers in parentheses refer to the References on page 253.

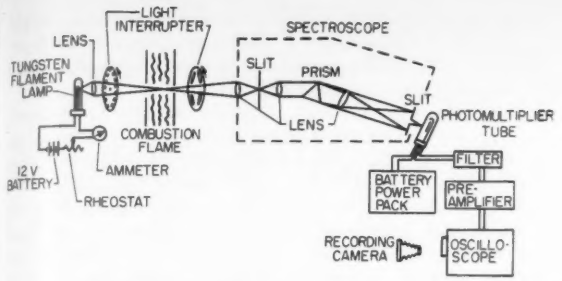

FIG. 1 SCHEMATIC DIAGRAM OF TEMPERATURE MEASURING APPARATUS

calibrated tungsten lamp, fixed wave-length spectroscope, mechanical light interrupters, photo multiplier tube, recording oscilloscope, and electronic component. A schematic arrangement of the instrumentation with relation to a combustion flame is shown in Fig. 1. In this figure, light from the tungsten lamp is shown to be periodically interrupted at a rate of 40 cps before passing through the combustion flame. The light from the combustor is also interrupted at a rate of 10 cps before entering the spectroscope. The photo multiplier tube is exposed to the light in a finite spectral band after passing through the spectroscope, and the resultant signal is amplified and recorded on photographic film by the recording oscilloscope.

The condition that must exist to obtain a temperature measurement from this arrangement is that radiation within the spectral band for which the spectroscope is adjusted must exist in the light originating in the combustion flame, and radiation over the entire band must exist in the light from the tungsten lamp. The radiation intensity of the lamp should be comparable to the intensity of the flame at the particular wave length; however, the two need not be equal. The particular spectral band selected for the investigation described in this paper centered about the sodium "D"-line. This band was selected because sodium is generally accepted as being in thermal equilibrium with the combustion flame, which is one of the necessary prerequisites for accurate temperature measurement. A sodium D-line of relative high intensity can also be readily produced in a flame by the addition of sodium salt to the fuel. Furthermore, the spectral radiation of the tungsten lamp is continuous throughout the sodium D-line region with comparatively high intensity.

Analysis of Temperature Signal

The recorded temperature signal obtained from a constant temperature flame with the described instrumentation is shown schematically in Fig. 2. The calibration run shown in this figure constitutes a record resulting from the radiation of the tungsten lamp through the combustion zone prior to actual combustion. With the optical system fixed and with constant amplification factors in the electronic components, combustion is started and the record for the temperature measurement run is obtained. In analyzing this signal the following nomenclature will be useful.

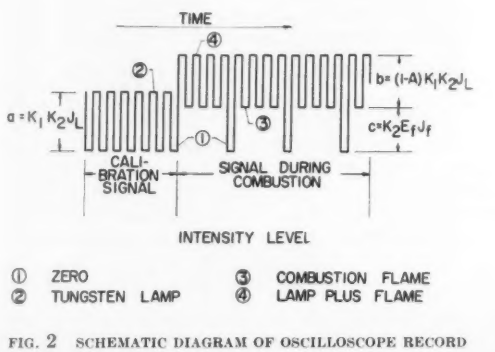

FIG. 2 SCHEMATIC DIAGRAM OF OSCILLOSCOPE RECORD

A_f = absorptivity of combustion flame
C_1 = known physical constant
C_2 = physical constant = $\frac{\text{Planck's constant} \times \text{velocity of light}}{\text{Boltzman's constant}} = 2.57$ cm °R
E_f = emisivity of combustion flame
J = radiation intensity at wave length λ
J_L = radiation intensity of tungsten lamp at wave length λ
J_f = black body radiation intensity for true flame temperature at wave length λ
K_1 = transmission coefficient between tungsten lamp and flame
K_2 = transmission coefficient between flame and film record
T_L = apparent temperature of tungsten lamp, °R
T_f = true temperature of combustion flame, °R
λ = wave length of light, cm

Referring to Fig. 2, the physical dimensions (*a*), (*b*), and (*c*) can be scaled from the oscillograph record. The quantity (*a*) is proportional to the radiation intensity of the tungsten lamp. It may be expressed as

$$a = K_1 K_2 J_L \quad [1]$$

The quantity (*b*) represents the amount of light transmitted through the flame and is the difference in radiation caused by the interruption of light from the tungsten lamp. When the light from the tungsten lamp is incident to the flame, the signal represents the radiation intensity of the flame plus the transmitted light from the tungsten lamp, both quantities being affected by the appropriate transmission coefficients. When no light is received from the tungsten lamp the signal represents only the radiation intensity of the flame which is also proportional to the dimension (*c*). In equation form the quantities (*b*) and (*c*) may be expressed as follows

$$b = (1 - A_f) K_1 K_2 J_L \quad [2]$$

$$c = K_2 E_f J_f \quad [3]$$

If the absorptivity and emissivity of the flame are equal (8), Equations [1] through [3] may be combined to obtain an expression for the ratio of the radiation intensity of the lamp to radiation intensity of the flame.

$$\frac{J_L}{J_f} = \frac{a - b}{K_1 c} \quad [4]$$

The radiation intensity for the black body is expressed by Wien's law as a function of temperature and wave length by the expression

$$J = C_1 \lambda^{-5} e^{-C_2/\lambda T} \quad [5]$$

Assuming this relation to be true for any radiating particle in thermal equilibrium, the following expressions are obtained

$$J_f = C_1 \lambda^{-5} e^{-C_2/\lambda T_f} \quad [6]$$

and

$$K_1 J_L = C_1 \lambda^{-5} e^{-C_2/\lambda T_L} \quad [7]$$

In Equation [7] the temperature of the lamp T_L is the temperature as seen through the optical system included in the constant K_1.

Combining Equations [4], [6], and [7], an equation relating the physical dimensions of the temperature record to temperature is obtained.

$$\frac{a - b}{c} = e^{C_2/\lambda (1/T_f - 1/T_L)} \quad [8]$$

The apparent flame emissivity may also be obtained from the

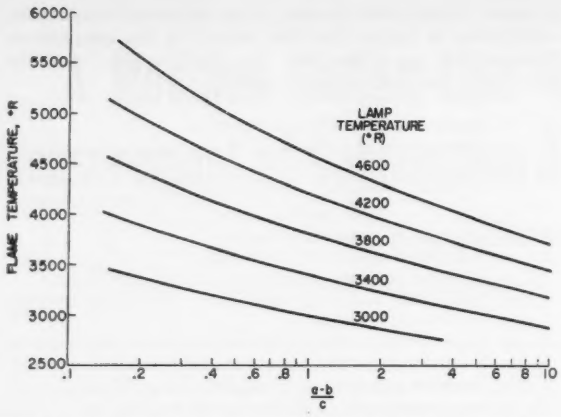

FIG. 3 VARIATION OF FLAME TEMPERATURE WITH THE QUANTITY $a - b/c$ FOR VARIOUS LAMP TEMPERATURES

temperature record. Combining Equations [1] and [2], an expression for emissivity and absorptivity is obtained.

$$E_f = \frac{a - b}{a} \quad \ldots\ldots\ldots\ldots\ldots\ldots [9]$$

In Equation [8] all quantities except flame temperature are known or measurable, hence flame temperature can readily be obtained.

Equation [8], for the average sodium D-line ($\lambda = 589 \times 10^{-7}$ cm), may be expressed as

$$\frac{a - b}{c} = e^{43.7(10^3/T_f - 10^3/T_L)} \quad \ldots\ldots\ldots\ldots [10]$$

A graphical representation of Equation [10] for several apparent lamp temperature is shown in Fig. 3. It can be seen from this figure that in calculating temperatures the variation in temperature is relatively insensitive to the quantity $a-b/c$ in the region of unity. The temperature variation is 16 to 23 per cent for a tenfold variation in the quantity $a-b/c$ over the range of lamp temperatures shown. Maximum accuracy is dependent primarily on the ability to accurately scale the dimension on the temperature record rather than the change in slope of the curves shown in Fig. 3. The optimum condition for accuracy in measuring occurs when the lamp temperature approximates the flame temperature and the flame has an emissivity of 0.2 or greater.

Experimental Investigation

Description of Apparatus

A rocket combustor designed specifically for the purpose of evaluating the usefulness of the temperature measuring technique was installed in one of the cells of the rocket laboratory at the NACA Cleveland laboratory. The installation was designed for simplicity of operation so that the major portion of the operating time could be spent developing the temperature measuring technique. A gas-pressurized flow system was used with individual flow controls for both propellants. Tank capacity was approximately two gallons with a maximum attainable flow of approximately 0.5 lb per sec. Spark ignition was used for all firings. An electric sequence timer was used to operate the flow valves required for engine firing. Liquid oxygen and fuel alcohol (95 per cent ethyl, 5 per cent methyl) were used as propellants. The liquid oxygen tank and flow line were cooled with liquid nitrogen during all firings.

The combustor installation and temperature measuring apparatus are shown in Figs. 4a and 4b. The combustor mounting, shown in the photograph, permitted height adjustment by motor drive. The adjustment allowed positioning of the various combustor stations in the optical path of the temperature measuring apparatus.

The configuration of the combustor and injector is shown in more detail in the drawing shown in Fig. 5. The combustor consisted of a tube 27 in. long and an internal diameter of 2 in. Temperature stations were located at $^3/_4$, $2^1/_4$, $6^1/_4$, $12^1/_4$ in. from the injector face. Air flow used to purge the windows of combustion gases was less than one per cent of the mass flow through the chamber. The injector was of the two-on-one inpingement type with a 90-deg included angle between the outer oxidant jets.

An 18-amp, 6-volt tungsten ribbon lamp was used as the comparison light source in the temperature measuring apparatus. A calibration curve of this lamp taken through the optical system preceding the midpoint in the combustor is shown in Fig. 6. The calibration was made with a standard optical pyrometer. Relative accuracy is believed to be good to within 20°R; however, absolute accuracy may be no better than 100°R. The experimental results are dependent primarily on relative accuracy and, therefore, the calibration was considered adequate for the particular application.

Experimental Procedure

Temperature variations were determined at various locations from the injector face for various oxidant-fuel ratios at

FIG. 4A TEMPERATURE MEASUREMENT APPARATUS AND COMBUSTOR INSTALLATION

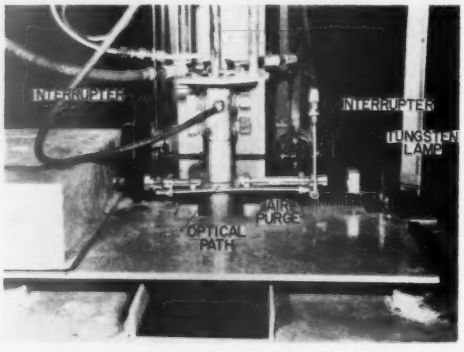

FIG. 4b TEMPERATURE MEASUREMENT APPARATUS AND COMBUSTOR INSTALLATION

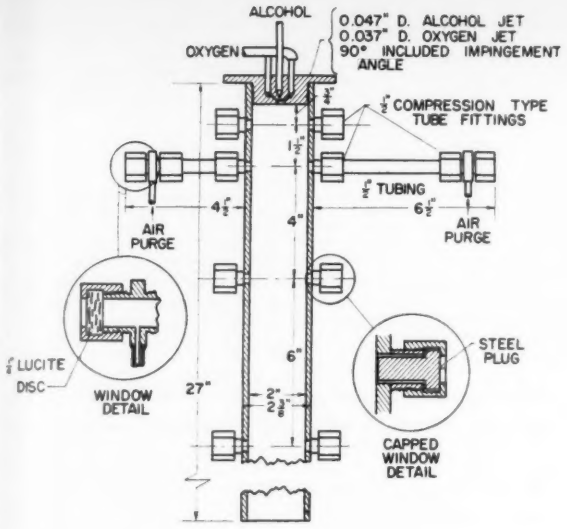

FIG. 5 COMBUSTION CHAMBER *abd* INJECTOR, SHOWING WINDOW DETAIL

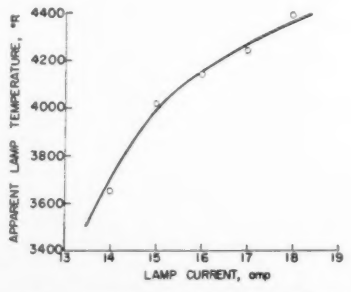

FIG. 6 TUNGSTEN LAMP CALIBRATION

a constant total weight flow. Water calibration of the injector indicated that a total propellant flow rate of 0.2 lb per sec provided an adequate range in O/F ratio without encountering extremely high or low injection pressure drops. Oxygen-fuel ratios of 0.8, 1.2, 1.6, 2.0, and 2.4 were selected as the five operating conditions to be used. The propellant tank-to-combustor pressure drop, weight flows, and calculated injection velocities for each test condition are listed in Table 1. These pressure drop conditions were calculated, using the coefficients from the water calibration and assuming a liquid oxygen density of 1.14 and an alcohol density of 0.78. All test runs were of approximately 6-second duration.

TABLE 1 OPERATING CONDITIONS

Weight flow ratio, Oxygen / Fuel	Pressure drop, lb/in.²		Calculated injection velocity, ft/sec		Weight flow lb/sec	
	Oxygen	Fuel	Oxygen	Fuel	Oxygen	Fuel
0.8	139	291	167	187	0.089	0.111
1.2	208	194	205	153	0.109	0.091
1.6	265	139	231	129	0.123	0.077
2.0	313	105	251	112	0.1334	0.0666
2.4	351	82	265	99	0.1412	0.0588

In order to obtain a strong sodium D-line in the combustion flame, a 2 per cent concentrated solution of sodium nitrite and water was used with alcohol. Solutions of sodium chloride and sodium iodide were also attempted and found adequate; however, the desired flame emissivity was experimentally evaluated with the sodium nitrite and was used for all the test conditions.

Results and Discussion

An oscillograph record obtained from a typical run with the experimental apparatus is shown in Fig. 7. The record obtained during combustion is preceded and followed by a calibration signal to insure that no changes occurred in the optical and electronic system during the run. A point-by-point analysis of a portion of this record indicates variations in temperature and flame emissivity at time intervals of $^1/_{40}$ sec as shown in Fig. 8. The average deviation in temperature is about 50 deg from a mean value with occasional deviations of 100 deg. Emissivity is shown to vary by 0.07. The actual variations in the temperature presumably exceeded those shown in Fig. 8 but were not recorded by the oscilloscope because of the electric filter network preceding the oscilloscope in the electronic system. Such a filter network was used because temperature variations exceeding the frequency of the light interruptions could not be evaluated. The recorded signal, therefore, represents an average condition over a finite time interval. More rapid temperature fluctuations can, however, be measured by this method by increasing the rate of interruption of light from the tungsten lamp.

The temperature defined by this measuring technique does not represent the average temperature in the optical path through the combustor. Rather, it approaches the maximum temperature occurring along the optical path through the combustor. A complete discussion of this characteristic is described in (9) and (10). This characteristic is both desirable and objectionable in measuring combustion temperatures. It is desirable in that cool boundary layers have relatively no effect on the measured temperature. The temperature measured is, therefore, indicative of conditions in the combustion zone itself. The weighting of the measured temperature toward the maximum temperature is undesirable because extreme temperature variations along the optical path result in a measured temperature that is not indicative of the average performance value.

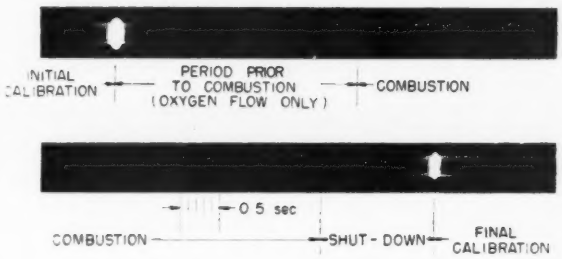

FIG. 7 TYPICAL OSCILLOSCOPE RECORD

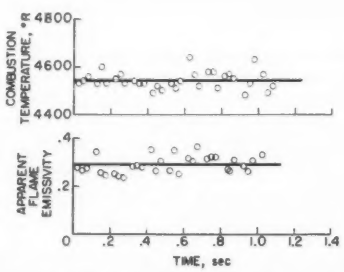

FIG. 8 VARIATION OF MEASURED COMBUSTION TEMPERATURE AND FLAME EMISSIVITY WITH TIME FOR TYPICAL FIRING

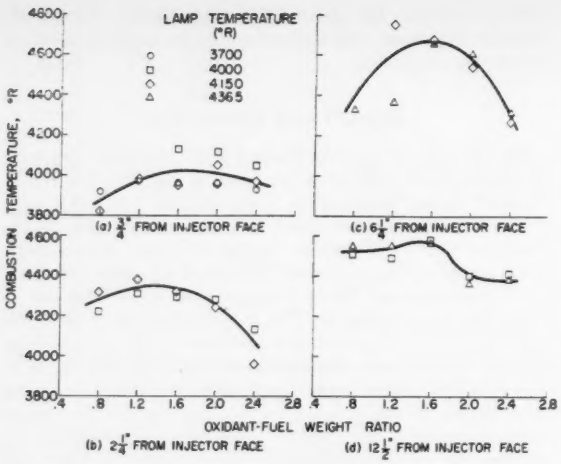

FIG. 9 VARIATION OF COMBUSTION TEMPERATURE WITH OXIDANT-FUEL RATIO AT FOUR COMBUSTOR STATIONS

(*a*) ¾ in. from injector face. (*b*) 2¼ in. from injector face. (*c*) 6¼ in. from injector face. (*d*) 12¼ in. from injector face.

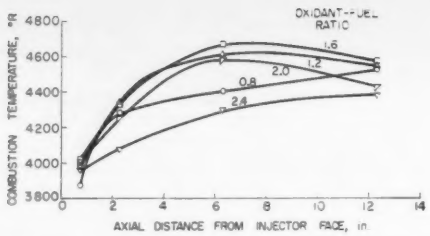

FIG. 10 VARIATION OF COMBUSTION TEMPERATURE WITH AXIAL DISTANCE FROM THE INJECTOR FACE AT VARIOUS OXIDANT-FUEL RATIOS. VALUES SHOWN WERE OBTAINED FROM CURVES IN FIG. 9

The measured temperatures for the various operating conditions are shown in Fig. 9. These temperatures are average values obtained over a 2-sec time interval. The temperature is shown as a function of oxidant-fuel weight ratio at the four stations. Comparatively good duplication of results were obtained for each condition. The temperature generally agreed to within 30 deg, which is the approximate accuracy of the measuring apparatus. Variations in excess of this amount are probably due to the inability to duplicate the operating condition. In addition to the accuracy of the propellant flow control apparatus, changes in propellant flow may have been incurred by changes in liquid oxygen density, resulting from helium absorption or improper cooling. Precautions were taken to minimize these effects; however, occasionally the effect may have been larger than anticipated. The lack of agreement in temperature at an oxidant-fuel ratio of 1.2 in Fig. 9 (*c*) may be the result of such changes. In general, however, the over-all accuracy of duplication was approximately 3 per cent. It is concluded from these results that the temperature measuring technique was accurate to within the accuracy of duplicating combustion conditions at a particular station.

Fig. 9 shows that the peak temperature did not occur at the same oxidant-fuel ratio at each combustor station. The peak theoretical temperature occurs at an oxidant-fuel ratio of approximately 1.9. The shifting of the peak temperature indicates a variation in the reaction time for each operating condition. This variation is more clearly shown in Fig. 10 where the variations in temperature are plotted as a function of the axial distance from the injector face. The general nature of the curves are as anticipated, in that the temperature increases with increasing distance from the injector face. However, a number of interesting characteristics are present.

One of the characteristics observed in Fig. 10 for oxygen-fuel ratios of 1.2, 1.6, and 2.0 is that the temperature decreased between the combustor stations located 6 and 12 in. from the injector. This could be the result of various factors. It is believed to be partly due to heat transfer. A heat-transfer rate of 0.25 Btu per sec per sq in. would result in a temperature decrease of approximately 100 deg. It may also result from mixing of high and low temperature gases since the measured temperature approximates the maximum temperature. Striated burning, therefore, will result in a measured temperature higher than the average temperature. A reduction in the measured temperature at a succeeding combustor station could result from the mixing of gasified propellants or low temperature products of combustion with previously measured high temperature gases. The continual increase in temperature with axial length at an oxidant-fuel ratio of 0.8 may have been influenced by such striated burning. At this operating condition the injection velocity of the alcohol was considereably greater than that of the oxygen, and, with the particular injector configuration used, it is probable that some of the alcohol penetrated a considerable distance downstream before burning. In fact, the striated appearance of the exhaust indicated the escape of unburnt alcohol at this oxygen-fuel ratio. Under such burning conditions the measured temperature is influenced by the volume of gas burning at a high temperature. With striated burning the volume of high temperature gases may increase with length and consequently produce an increase in measured temperature.

Although the theoretical temperature variation with oxidant-fuel ratio for the pressure condition used in this investigation was not available, temperature variations of more than 1000 deg were anticipated between an oxidant-fuel ratio of 0.8 and 1.9. Using the theoretical temperature variation at higher chamber pressures as a reference, such a difference is indicated. The measured temperature, however, varied by no more than 500 deg at any station over the oxidant-fuel ratio range, with the measured temperature at fuel rich mixtures being higher than was anticipated. This may be the result of striated burning as previously stated, and the small spread in temperature suggests that burning occurs at or near the same oxidant-fuel ratio independent of the operating condition. Such an explanation does not preclude the occurrence of other phenomena. A combustion temperature at fuel-rich mixtures that is higher than theoretical calculations has been reported for some combustible mixtures (11). The reaction of sodium with combustion products may also influence the measured temperature under certain conditions (12).

The apparent emissivity of the combustion flame is shown as a function of oxidant-fuel ratio for each combustor station in Fig. 11 and as a function of axial length in Fig. 12. The apparent emissivity is important primarily with regard to obtaining adequate variations in recorded radiation intensity levels. Variations, however, may be significant in predicting changes in combustion. Fig. 11 indicates that emissivity remained relatively constant at each station with all oxidant fuel ratios. At several conditions emissivity appears to vary with lamp temperature. The conditions of constant lamp temperature at each station were obtained in successive firings, and the change in emissivity may be the result of an unpredictable change in sodium content of the fuel. The change in emissivity with axial distance from the injector face (Fig. 12) appears to be more significant. The emissivity is shown to increase rapidly as the injector is approached. The temperature data indicate that the primary reaction zone occurs within the first 6 in. from the injector, which may be the cause of the higher flame emissivity.

One factor influencing the apparent flame emissivity that requires further investigation is the effect of the liquid propellant interference on the transparency of the flame at the so-

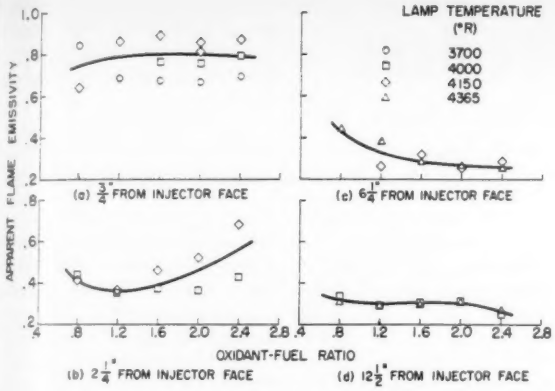

FIG. 11 VARIATION OF APPARENT EMISSIVITY WITH OXIDANT-FUEL RATIO AT FOUR COMBUSTION STATIONS

(*a*) 3/4 in. from injector face. (*b*) 2 1/4 in. from injector face. (*c*) 6 1/4 in. from injector face. (*d*) 12 1/4 in. from injector face.

dium wave length. This factor would increase as the injector is approached and could result in an increase in measured emissivity. The effect is not believed to be large; however, it could not be evaluated with the available apparatus. The effect of a high emissivity would result in a measured temperature that is too low. If this effect is present, the general nature of the temperature variation with axial length (Fig. 10) would remain essentially the same except that the initial increase in temperature would be less rapid.

Conclusions

The present investigation indicates that the modified sodium-line reversal technique or similar methods utilizing other spectral bands may be successfully applied to the measurement of relative temperatures at various stations in a rocket combustor. The experimental results obtained with an open tube rocket combustor show that the axial temperature distribution is dependent on the operating condition. These data appear to be valuable in describing the influence of the injection process on combustion performance.

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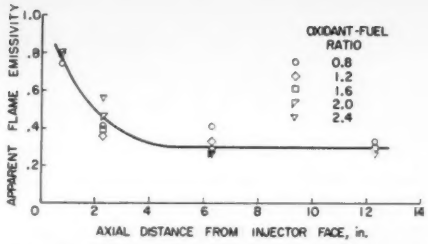

FIG. 12 VARIATION OF APPARENT FLAME EMISSIVITY WITH AXIAL DISTANCE FROM INJECTOR FACE FOR VARIOUS OXIDANT-FUEL RATIOS. VALUES SHOWN WERE OBTAINED FROM CURVES IN FIG. 11

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Technical Notes

EDITOR'S NOTE: This section of the JOURNAL is open to short manuscripts describing new developments or offering comments on papers previously published. Such manuscripts are published without editorial review, usually within two months of the date of receipt. Requirements as to style are the same as for regular contributions. (See second page of this issue.)

Jet Propulsion News

C. F. WARNER, Purdue University, *Associate Editor*

with the assistance of W. G. BOHL

Guided Missiles and Rockets

THE U. S. Defense Department has released limited information concerning three of this country's guided missiles, the Regulus, the Nike, and the Bomarc.

The Regulus (Figs. 1 and 2) was developed by Chance Vought Aircraft Division of United Aircraft Corporation under sponsorship of the Navy Bureau of Aeronautics for launching from submarines, surface ships, and shore bases. Launching equipment can be installed in a short period of time on several types of vessels at relatively low cost and with only slight modifications of the ship itself.

The U. S. submarine Tunny, recommissioned on the West Coast, March 6, has been specifically modified to launch the Regulus. The Tunny is a converted World War II submarine that has been modernized by the addition of the snorkel and streamlining of the hull and conning tower. While in the shipyard, a tank for stowing a guided missle and a launching rack was installed. A small group of officers and enlisted men on the Tunny have been specially trained for the past year at the U. S. Naval Air Missile Test Center at Pt. Mugu, Calif., in the operation and maintenance of the Regulus.

Although the assault missile, and certain other configurations, will employ a drone version of Regulus, tactical employment will also include those techniques and guidance systems associated with the operation of all-weather, distantly controlled guided missiles. Such plans will make it possible to utilize the missile in various ways without the expense and effort of designing and procuring a separate missile for each function.

The test and training versions of the Regulus missile are equipped with tricycle landing gear so that the missile may be recovered upon completion of its flight. This recovery feature is important because the missile is not lost after each flight. A flight test vehicle, during the early stages of development, approximates the cost of a jet fighter. To evaluate a jet fighter, approximately 100 hr of flight time are required. To obtain the same flight test information on a nonrecoverable missile comparable to the Regulus, around 200 missiles would have to be used if each were expended. The recovery feature permits the number to come down to about 30.

In addition, much important test data, which might be lost if the missile were destroyed, are recovered and used to good advantage in subsequent flights. Several test missiles now in use have been flown many times to subsonic and supersonic speeds and have been recovered without damage. The Navy states that as many as 10 flights have been made with a single vehicle, cutting to one tenth the cost of a comparable operation involving loss of a vehicle or missile for each test. Experience in landing the Regulus has shown that it may be recovered through the use of a parachute-type brake on an average-sized runway.

The tactical missile has no landing gear but carries a powerful warhead. In appearance, Regulus resembles a conventional, swept-wing jet fighter. It is about 30 ft in length.

Service plans envisage a wide use of the Regulus missile both as tactical weapons and as recoverable test vehicles. Tactically, Regulus will be used against appropriate land targets and in amphibious warfare by the U. S. Marine Corps and the Navy. The recoverable version is being used to train operating units in launching and guidance techniques and can be used as a high-speed drone for antiaircraft guided missiles and antiaircraft gunnery.

Now in production at Chance Vought's Dallas, Tex., plant, Regulus has completed the first phase of its operational tests in a completely satisfactory manner, and performance has exceeded early design specifications. The missile was ini-

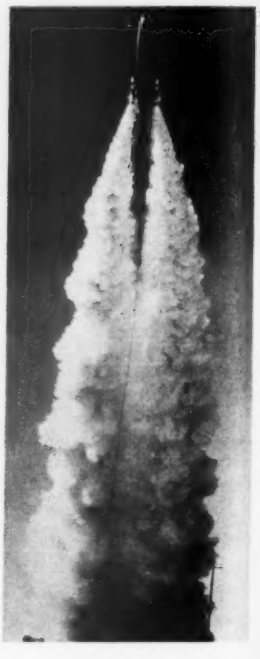

FIG. 1

THE REGULUS BEING LAUNCHED BY TWO BOOSTER ROCKETS. THE LAUNCHING CONTROL CABLE CAN BE SEEN STILL ATTACHED TO THE REGULUS

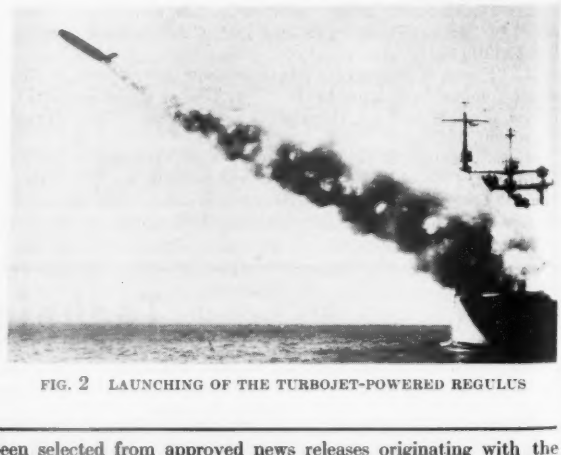

FIG. 2 LAUNCHING OF THE TURBOJET-POWERED REGULUS

EDITOR'S NOTE: The information reported in this Section has been selected from approved news releases originating with the Department of Defense, private manufacturers, universities, etc., and from published news accounts in journals and newspapers. The reports are considered generally reliable, although no attempt has been made to verify them in detail.

tially developed in 1948, and Regulus was first flown at Edwards Air Force Base in 1950.

The Nike, developed by Douglas and Bell Telephone under sponsorship of the Army, is designed for ground-to-air use in defense of U. S. cities. This target-seeking missile has proved effective at altitudes above 75,000 ft and at ranges of about 35 miles. It is 20 ft long and about 1 ft in diameter. The Nike's approximate 1000-lb weight is launched by a solid propellant rocket and powered in flight by a liquid fuel rocket. The Nike is currently undergoing flight tests at White Sands Proving Ground.

The supersonic Boeing-developed Bomarc is a rocket- and ramjet-powered guided missile capable of banking and turning in pursuit of its target. It is designed for ground-to-air use. The inboard rocket power plant is used to launch the Bomarc and to bring it up to its supersonic speed at which time the ramjets take over. The Bomarc, an outgrowth of Boeing's GAPA experimental program is now in large-scale production.

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ITALJET (Compagnia General Italjet) is now delivering 14-AS-1000 Jato units to the United States Air Forces in Europe. Italjet will also manufacture bazookas for the North Atlantic Treaty Nations.

✦ ✦ ✦

IT WAS learned that an SEPR rocket-powered missile built by Société Matra has been successfully tested in level flight at a speed of 1100 mph at the French Air Force test range in North Africa. The power plant, produced by Société Pour l'Étude de la Propulsion par Reaction, is said to produce 2750 lb thrust for 14 sec. The all-metal missile, known as the M.04, is said to have a length of approximately 15 ft and a diameter of 16 in. The empty weight is 770 lb but the take-off weight is supposed to be 1020 lb. No other details are known.

✦ ✦ ✦

LOS ANGELES Airways and Aerojet General Corporation are currently working on a joint program to equip the rotor blades of the Sikorsky S-55 helicopter with rocket motors. The rocket motors designed for emergency use will provide sufficient power for increased performance and safety.

✦ ✦ ✦

THE U. S. Air Force recently unveiled a hitherto secret rocket-firing device in the North American F-86 D Sabre Jet interceptor which automatically releases 24 Mighty Mouse 2.75-in. rockets in a single volley (Fig. 3). The

FIG. 3 SABRE JET'S NEW ROCKET-FIRING POD

rockets are launched from a retractable pod that snaps back into the plane's fuselage immediately after firing. The rockets, each equivalent to a 75-mm artillery shell in hitting power, leave the plane with a velocity of approximately 200 mph.

Aircraft

THE North American Aviation AJ-2 Savage, advanced version of the carrier-based AJ-1 U. S Navy bomber, has been undergoing flight tests. Capable of delivering an atomic bomb from either carrier or land bases, the three-engined, 25-ton AJ-2 is the Navy's largest carrier-based bomber. The Savage is powered by two Pratt and Whitney R-2800 reciprocating engines and an Allison J-33 turbojet. All engines burn a common fuel. The Savage is expected to have a speed of approximately 425 mph.

✦ ✦ ✦

THE Bell rocket-powered X-1A, latest in the X-1 research aircraft design series, is now undergoing initial flight tests at the Air Force Flight Test Center, Edwards, Calif. The X-1A is larger than the X-1 and has a more powerful Reaction Motors-built liquid-oxygen and alcohol-water rocket motor. The X-1A (Fig. 4) is expected to have a top speed of Mach 2.5,

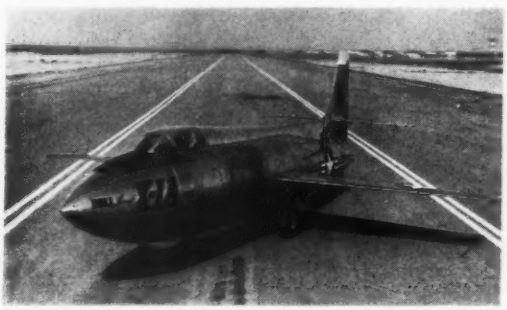

FIG. 4 BELL AIRCRAFT X-1A RESEARCH PLANE

It is understood that Bell Aircraft Corporation may produce a vertical take-off rocket-powered interceptor. The plane would be capable of taking off vertically under its own power without the use of launching racks or guides and would also be able to land in the same area the size of a tennis court.

✦ ✦ ✦

A BELL Aircraft B-50 bomber carrying the experimental Bell X-2 exploded at 30,000-ft altitude over Lake Ontario recently. The explosion forced the crew of the "mother" bomber to jettison the highly secret rocket plane which dropped into the lake about 15 miles from Rochester, N. Y. A Bell spokesman said the explosion was aboard the mother ship. But he explained that the X-2 was damaged and that the complicated mechanisms of carrying a "parasite" plane made it difficult to be certain that the X-2 itself did not explode.

✦ ✦ ✦

BOEING Airplane Company is currently producing a reconnaissance version of the Boeing B-47 Stratojet bomber for the U. S. Air Force. The high-speed, high-altitude plane will carry a three-man crew.

✦ ✦ ✦

ACCORDING to George S. Schairer of Boeing Airplane Company, the use of jet engine pod power plant mounting on multi-engine aircraft instead of the conventional submerged mountings should result in better over-all aircraft performance. Additional drag brought about by the pod is offset to some extent by the thinner wing section which is not required to house a submerged engine. Some loss in engine power in submerged installations caused by the need to channel inlet air and exhaust gas through long ducts is all but eliminated in the pod installation. The pod-mounted engine is freer from fire hazards and more accessible for maintenance.

✦ ✦ ✦

THE rocket-powered SO 9000 Trident supersonic flight research plane is being flight-tested in France. The SO 9000 is similar in appearance to the Bell X-1 and is currently

being powered by two wing-tip-mounted Turbomeca Marbare turbojet engines. A second model of the Trident is supposed to fly this coming autumn.

✦ ✦ ✦

CONVAIR is reported to be designing a delta-wing supersonic bomber designated the XB-58. Four Pratt and Whitney J57 engines are understood to power the plane at a speed of about Mach 1.

✦ ✦ ✦

LOCKHEED Aircraft Corporation is equipping its F-94C Starfire with deceleration chutes for the U. S. Air Force. Studies have shown that the use of the chute cuts down the Starfire's required landing distance by one half and doubles the tire and brake life (Fig. 5).

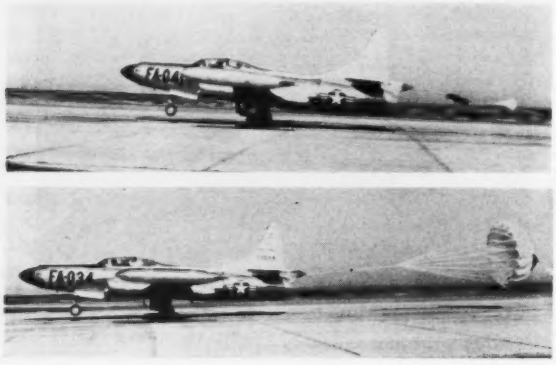

FIG. 5 STARFIRE DECELERATION CHUTE

Split-second action of the slow-down parachute released from the tail of a Lockheed F-94C Starfire, USAF newest all-weather radar-equipped jet interceptor, is demonstrated in these photos taken during tests at a California base. The sequence shows one plane touching down while the small pilot chute pulls a "package" containing the full-sized tight-reefed drag parachute from its special storage compartment above the tailpipe. In the lower picture, showing another plane, the 16-ft diam deceleration chute has popped open and immediately begins to slow down the F-94C. The 600-mph-plus Starfire is the first American production fighter aircraft to have a drag parachute installation

✦ ✦ ✦

THE Air Force has announced the successful flight of the new Sabre Jet, the F-86 H. The completely new jet, shown in Fig. 6, is slightly larger than current Sabres and is powered by the new General Electric J-73 turbojet engine.

Turbojet Engines

THE Air Force has notified the General Electric Company that a tremendous increase in the life expectancy of General Electric J-47 engines will save the American taxpayers over one hundred million dollars. Air Force sources in Washington revealed that a 100 per cent increase in service time between overhauls has been realized on General Electric's J47-25 jet engines which power the Boeing B-47 Stratojet bomber. The increase in operating time between overhauls will reduce the number of engines required to support the B-47 program, it was revealed. Consequently, the procurement of J47-25 engines for B-47 aircraft has been reduced, which will result in a saving of over one hundred million dollars.

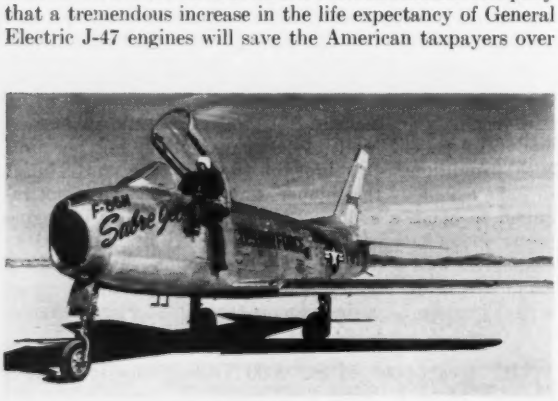

FIG. 6 F-86H SABRE JET

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THE Olympus-powered Canberra which was shown at Farnborough last year has for some time been carrying out high-altitude flying as part of the normal development of the engine for military use, and has attained heights of over 60,000 ft. This is of outstanding operational importance, as is also the remarkably consistent performance up to these very great altitudes. No decision has yet been taken about an attempt on the official height record of 59,446 ft, but consideration will be given to this if the attempt can be made without interference with the main program of engine development. The Olympus is the only two-shaft turbine flying in the country and indeed, apart from the American J57, the only one in the world.

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THE lifting of secrecy surrounding the Ryan Firebee has created considerable interest in its power plant, the Fairchild J-44 turbojet engine. The J-44, described as the most powerful engine of its size, delivers 1000 lb of thrust with a weight of only 300 lb. It is a compact cylindrical package only 72 in. long and 22 in. in diameter. The engine is a self-contained power plant, having its own oil storage, needs no external oil coolers and, because of its smooth outer shell, needs no additional cowling. The J-44 is said to be extremely easy to service.

✦ ✦ ✦

SHIRO KAYABA, a leading Japanese industrialist, recently paid a visit to Boeing Airplane Company's gas turbine project for a look at the Boeing-developed Model 502 turbine engine and information on its application to commercial helicopters. Kayaba, chairman of the board of Kayaba Industry Company, reports that Japanese industry is vitally interested in the turbine. He is scheduled to report on it at a meeting of Japanese manufacturers and engineers on his return.

Following his visit to Boeing he will tour other American industrial plants and military bases.

New Facilities

THE merger of Aerojet Engineering Corporation and the Crosley Motors, Inc., of Cincinnati, Ohio, has been announced by Mr. A. H. Rude of Aerojet. This merger will provide much needed metal parts fabricating facilities. The new organization will be known as the Aerojet General Corporation.

✦ ✦ ✦

TWO years of intensive research and development work at Vickers, Inc., in cooperation with the Navy Bureau of Aeronautics, have resulted in the creation of the universal turbojet engine simulator for developing, testing, and calibrating fuel control components for gas turbine engines.

Preliminary results indicate that the cost of final calibration of fuel controls on an engine can be reduced to about 2 per cent of their present cost by use of this engine simulator. The testing of a turbojet-engine fuel control has in the past involved little more than steady-state operation of the control at a number of discrete speeds within its operating range. This static type of test is at best a poor measure of whether the fuel control will operate satisfactorily with a particular engine.

A given fuel control system, when coupled to an engine, may induce unstable operation even though the foregoing bench tests results are outwardly satisfactory. This apparent paradox can arise from an improper time constant in the fuel regulating system, or any of the several dynamic effects in the control that are not revealed by the comparatively crude bench tests.

A further limitation of most of the conventional fuel control test stands is their inability to maintain the speed of the control in the governing range of operation. This is a natural limitation originating in the negative power slope characteristics of the governing curves, but one which can be overcome with a properly designed test stand.

A single-engine simulator is readily adaptable to testing fuel systems for all current gas turbine engines including (1) turbo-propeller engines, (2) turbojet engines with fixed area tail, (3) turbojet engines with afterburner, (4) turbojet engines with variable area.

With this device it will be possible for engine manufacturers to have a newly designed fuel system thoroughly tested and developed prior to its installation on a prototype engine. This procedure should eliminate many costly months presently devoted to conducting fuel-system development work on experimental engines.

✦ ✦ ✦

DEVELOPMENT testing of various types of jet propulsion units, used to power several types of helicopters now under development at McDonnell Aircraft, is scheduled to begin in a recently completed $1,200,000 propulsion test laboratory during April. The new lab, located on Brown Road about one mile east of the main McDonnell plant, is now undergoing final acceptance tests.

Strong blasts of air, with a velocity of hundreds of miles per hour, will be forced through two test chambers in the lab. In these chambers, jet propulsion units will be ignited and operated under simulated flight conditions, and other items of jet aircraft, helicopters, and missiles will be tested. The tremendous air flow will be produced by two compressors, which can be run alternately, from the power provided by a 3000-hp motor.

Because some propulsion units are noisy in operation, extra precautions in construction of the laboratory have been taken to deaden the sound. This is done with 12 in. of reinforced concrete, and 4 in. of soundproof insulation plus a "sinusoidal labyrinth" type of construction in the sound chambers. The labyrinth's twisting passages between the concrete walls permits exhaust and steam from the jets to escape outside while the noise is retained within.

To illustrate how efficiently sound is deadened in these passages, a person shouting at the top of his lungs at one end of the "labyrinth" can't be heard by another person standing 25 ft away at the other end of the passages. In the same way, the noise of jet propulsion units in operation is only a murmur outside the laboratory.

Two control rooms, one for each compressor, are of "floating cell construction" to minimize noise and vibration which might interfere with measuring instruments in the control rooms.

The compressors themselves, representing $850,000 of the cost of the lab, are especially designed for specific development tests. One is a relatively high-pressure low-mass-flow compressor, and the other will deliver a $13^1/_2$-in. diam jet of air at near sonic velocity. The second compressor is for use in developing subsonic ramjets, testing air inlets, small transonic wind-tunnel models, ram-air instruments, and helicopter applications.

✦ ✦ ✦

LOCKHEED Aircraft Corporation reported awarding of contracts approximating $3,200,000 for construction of three factory buildings at the U. S. Air Force's new jet test center at Palmdale.

Planned for utilization in Lockheed's present and long-range activities at the desert site, the three new structures include: a 180 by 340-ft final assembly building, a 150 by 675-ft production flight hangar, and an 82 by 160-ft maintenance and service building. Envisioned as ultimately becoming the world's largest jet flight center, the manufacturing and flight test facilities at Palmdale will be shared by three major aircraft companies—Lockheed Aircraft Corporation, North American Aviation, Inc., and Northrop Aircraft, Inc.

✦ ✦ ✦

TWO of the new Giant 11-ton electronic calculators recently unveiled by IBM and featured in the current issue of *Collier's* will be installed by Douglas Aircraft Company in its Santa Monica and El Segundo plants in the near future. In just six minutes these monster computers, for whose use Douglas will pay a monthly rental of $15,000 each, can solve mathematical problems that one man with a desk calculator would require eight years to do.

✦ ✦ ✦

CREATION of a new Engineering Division of Marquardt Aircraft Company giving a more independent status to airborne accessories for piloted aircraft and guided missiles was announced by Roy Marquardt, president. The new division will be known as Accessories Engineering with Robert L. Goodyear as chief engineer, while the present engineering unit will be designated Engine and Afterburner Engineering with John A. Drake continuing as chief engineer.

✦ ✦ ✦

A WIND tunnel E-1 capable of speeds between 1000 and 3500 mph has been completed at the Air Force's Arnold Engineering Development Center, Tullahoma, Tenn. The intermittent supersonic tunnel is one of several small tunnels included in the Gas Dynamics Facility. The total cost of this part of the Gas Dynamics Facility is reported to be $470,000.

Personalities

FURTHER steps in establishing Boeing Airplane Company's pilotless aircraft program on a divisional status were made by the appointment of five new men to the program. The newly assigned engineers are: R. E. BOTHELL, LEWIS DANIELS, REX DELAWARE, CLYDE PETERSON, and RICHARD NELSON.

✦ ✦ ✦

LT. COL. JOHN P. STAPP has been appointed as Chief of the Space Biology Laboratory at the Holloman Air Development Center, Alamogordo, N. Mex. Col. Stapp will be in charge of such research projects as air-crew ejection escape from all types of aircraft, human wind blast tolerances up to 800 mph, biological effects of cosmic radiation at high altitudes, and effect of high-altitude research rocket flights on mammals.

✦ ✦ ✦

FRANCIS J. GAFFNEY has been appointed to the post of director of engineering for the Guided Missiles Division of the Fairchild Engine and Airplane Corporation. Mr. Gaffney will have broad responsibility for all phases of the engineering program of the Division.

✦ ✦ ✦

L. EUGENE ROOT has been chosen as Director of Development Planning for Lockheed Airplane Corporation. Root and his group will study directions of aviation development, trends in power, speed, and aircraft capacity; the utilization of scientific knowledge; and will forecast generally corporate activities in future weapons systems involving new types of airplanes and missiles.

American Rocket Society News

H. K. WILGUS, *Associate Editor*

Newly Formed ARS Arizona Section Receives Charter

THE charter officially establishing the Arizona Section of the American Rocket Society was received by the Arizona group at a Charter Presentation Dinner held at the Tucson, Ariz., Elks' Club, Friday evening, May 1, 1953. One hundred and fifty members and guests attended to witness the presentation and to hear the two guest speakers, Lt. Col. W. P. Patterson, Chief of Technical Staff, White Sands Proving Ground, and Edward E. Francisco, Jr., Head of the Propulsion Branch, WSPG. Mr Francisco is also president of the ARS New Mexico-West Texas Section.

The charter was presented to the 70 members and its seven officers in behalf of the national society by Edwin J. Burnell, mechanical engineer of the Hughes Aircraft Company, Culver City, Calif. Anthony R. Tocco, first president of the Arizona Section, received the charter.

WSPG Activities Described by Patterson

Colonel Patterson, the first guest speaker, for many years interested in rocketry and its role in national defense, spoke on the nature and operation of the WSPG in New Mexico. Unable to reveal technical details, Col. Patterson did, however, tell many interesting facts about experimental work going on at WSPG, and said that the firing range was a huge physics laboratory where electronic and optical instruments are scattered over an area of 4000 square miles.

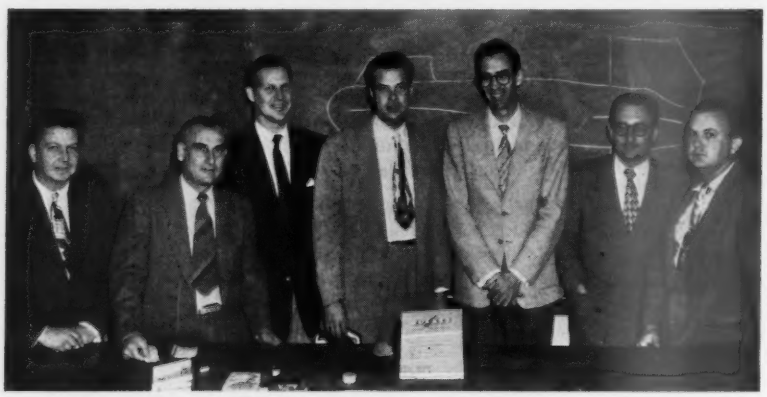
During his lecture, Col. Patterson projected colored slides showing some of the measuring instruments, cameras, and test equipment now in use at WSPG.

Origin of ARS Told

Mr. Francisco spoke on the origin of the American Rocket Society which, he said, grew out of the earlier American Interplanetary Society composed mainly of a group of science fiction writers who met in small informal groups to discuss space flight and possible life on other planets. He also told of the formation of the New Mexico-West Texas Section, of which he is president. He expressed amazement at the way in which the Arizona Section mushroomed to its present size of over 70 members since the first organizational meeting held in January 1953.

The larger part of the Arizona group is comprised of Hughes Aircraft Company employees, but also included are personnel of the Grand Central Aircraft Company, Davis-Monthan Air Force Base, faculty members and students of the University of Arizona, and other professional and technical people in the Tucson-Nogales area.

The Arizona Section Board of Directors, in addition to Mr. Tocco, is composed of Dr. John Anthony, geologist at the College of Mines, University of Arizona; Robert F. Buelow, Jack D. Clothier, and Jack W. Biggam, all of Grand Central Aircraft; Edward S. Forman, explosives engineer; and James Chinello, technical editor.



BOARD OF DIRECTORS OF THE NEWLY FORMED ARIZONA SECTION, AT THE FIRST BOARD MEETING, MARCH 11, 1953

Left to right: Jack W. Biggam, membership chairman; James Chinello, historian and correspondent; Robert F. Buelow, finance chairman; Edward S. Forman, secretary-treasurer; John Anthony, vice-president; Anthony R. Tocco, president; and Jack D. Clothier, programs chairman

Norton Williams of Vickers, Inc., Addresses New York Section

ON March 20, 1953, members and guests of the ARS New York Section met at the Engineering Societies Building, 29 West 39th Street, New York, N. Y., to hear Norton Williams of Vickers, Inc., present a talk on "The Use of Hydraulic Components in Servo Circuits." The following is a summary of his remarks.

Servo theory is having a very important influence on the design of pumps, motors, and valves in the field of modern fluid mechanics. To the usual specifications for hydraulic mechanisms (pressure, flow, speed, static response, etc.) servo engineers now require additional factors, such as: (1) Dynamic performance in the form of phase and amplitude response to both sinusoidal and step-input signals; (2) effect on dynamic performance of various feed-back control circuits; (3) torque-squared-to-inertia ratio; (4) time constant; (5) damping factor; (6) integration with electrical transducers and electrical control circuits to form electro-hydraulic servo systems.

This situation is having the healthy effect of stimulating hydraulic equipment manufacturers to be much more cognizant of these factors, and to take steps to improve them. The general result of this improvement is the increasing use of hydraulics in projects requiring high-response and high-accuracy control and actuation. One of many application examples is electro-hydraulics as employed in various control functions on rockets, rocket launchers, missiles, and jet engines.

A specific example of a modern hydraulic mechanism is the Servo Pump Unit, whose amount and direction of output flow can be varied at will, and which is specifically designed for servo circuit application. The combination of this pump and a rotary or linear, hydraulic motor forms an electrically controlled hydraulic transmission. This type of transmission may be considered a power amplifier when viewed from the electrical signal input of about five watts to the mechanical power output of several thousand watts. Pressure drop across the ports of a metering valve, with its inherent losses, is avoided in the main power system by confining metering valve action to a low power level system. This is the stroking system. It is a part of the servo pump unit. Therefore, output from the power pump is not dependent upon pressure-drop methods of control, but is determined by the amount of flow which the stroking system directs the power pump to develop, and by the actual resistance offered by the load. In a control system employing a servo pump, variations in gain resulting from a change of load are negligible, as compared to those which may exist in a similar circuit whose control is exercised by a valve metering directly in the main power line.

In summarizing, Mr. Williams stated: (1) Modern fluid mechanics offers mechanisms of exceptional suitability as the power transmission and associated controls in feed-back type systems. (2) This is due to

basic inherent characteristics of hydraulic mechanisms and circuits, and to the lively impact which servo theory is now having on the design of hydraulic mechanisms.

Following the Section meeting, the audience stayed for more than an hour to view a colored film entitled "Navy and Science," and to inspect an interesting display of mechanical parts.

New Mexico-West Texas Section Hears Wagner on WSPG Range Safety

A LARGE audience gathered to hear Nathan Wagner, Chief of the Missile Flight Safety and Range Control Division, WSPG, speak before the New Mexico-West Texas Section meeting on March 26, 1953, at the New Mexico College of Agriculture and Mechanic Arts, Las Cruces, N. Mex.

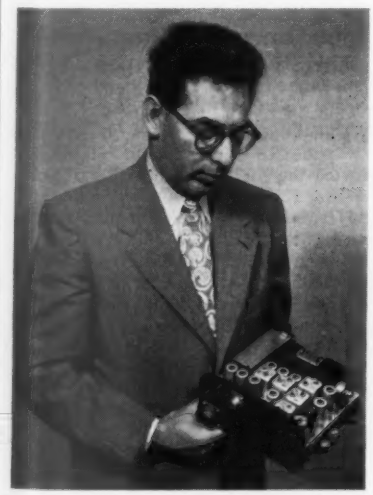
Mr. Wagner's topic was "White Sands Proving Ground Range Safety." In setting up a range safety program, he stated, it is necessary to meet certain basic requirements with regard to a special missile. Each type of missile must be treated separately, since missile flight characteristics differ; e.g., thrust, drag, burning time, velocity at burn-out, etc. The calculated missile characteristics and flight path must be known, and the minimum deviation from the proposed flight path which would endanger the range boundaries must be calculated. The information is then plotted as a continuous curve or set of curves on a plotting board with the safety lines included. Some reliable means must be at hand for "cutting down" the missile either by shutting off the motor or by breaking up the missile and causing it to become aerodynamically unstable.

Once the initial conditions have been set up, Mr. Wagner continued, the particular missile project officer is required to provide all information of tests which will have any bearing on the normal performance of the missile, and to include all data on the proposed flight path and calculated trajectory. Using this information, a family of curves are calculated, assuming missile failure at various times after take-off. From these curves, calculations are made to determine how long a missile may be allowed to continue under thrust without violating the range boundaries. All range safety operations must be performed during the powered portion of the flight. These points are plotted on a range map with a curve drawn through the points. This curve is known as the safety line. If a missile crosses this line under power, it is unsafe and must be cut down.

At White Sands, radars are used to track the missile in the air. The results of this tracking are presented as a continuous curve on the plotting board. From this information, the Flight Safety Director can determine the relative position of the missile and decide whether it is safe or not. If the curve passes over the safety line, the missile may violate the range boundaries.

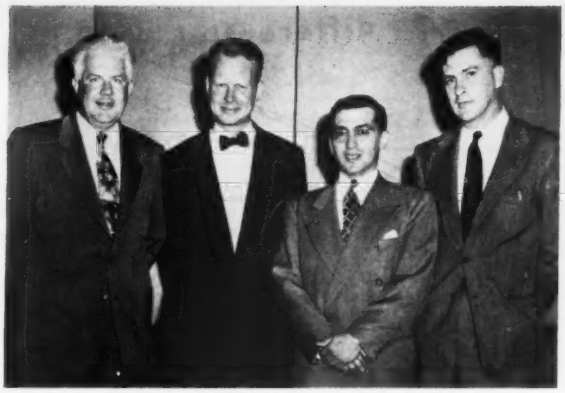
However, Mr. Wagner remarked, all the information and calculations described are useless unless some reliable means are available to effect the missile cutoff. Two systems are used at WSPG, he said. By changing the pulse repetition rate of the radars tracking the missile, the missile's radar beacon may be made to actuate the cut-off mechanism in the missile. Or, using the second method, a radio receiver may be used to perform the same function. This receiver is a small, compact intelligence receiver which closes relays corresponding to the transmitted command. This starts a chain of events which performs either fuel shutoff, missile separation, or missile destruction.

While no one is infallible, the record of the Safety Branch at WSPG is clear to the present time, Mr. Wagner pointed out. There have been no failures to terminate a flight when required, and no known failures of the safety equipment while a missile was in flight.



MR. NATHAN WAGNER, GUEST SPEAKER AT THE NEW MEXICO-WEST TEXAS SECTION MEETING ON MARCH 26, 1953

Mr. Wagner, who is chief of the Missile Flight Safety and Range Control Divisions, WSPS, is showing one of the compact radio units installed in a missile and used to actuate elements for shutting off or destroying unsafe missiles in the air over WSPG



AT THE NORTHEASTERN NEW YORK SECTION DINNER MEETING MAY 12, 1953

Left to right: Dr. R. W. Porter, ARS Board of Directors; Mr. F. C. Durant III, ARS president; Dr. Kurt Berman, ARS Board of Directors; and George E. Moore, president of the Section

Northeastern New York Section Announces 1953 Officers

THE following officers and Board of Directors for 1953 of the ARS Northeastern New York Section were elected at a recent meeting:

President, George E. Moore, Research Laboratory, General Electric Company, Schenectady, N. Y.; *Vice-President*, Elliot Ring, Malta Test Station, Ballston Spa, N. Y.; *Treasurer*, Harold M. Weber, Malta Test Station; *Secretary*, Theodore C. Carnavos, Malta Test Station.

The Board of Directors consists of the following: Edwin H. Hull, Malta Test Station; Joseph C. Hoffman, Campbell Avenue Plant, GE, Schenectady; George J. Mullaney, Research Laboratory, GE, Schenectady.

The Section announces that its new official mailing address is: Malta Test Station, General Electric Company, Ballston Spa, N. Y.

Durant Discusses Rocket Power

Mr. F. C. Durant, III, President of the American Rocket Society, was the guest speaker at the Section's Annual Dinner Meeting held on May 12 at the Edison Club, Schenectady, N. Y.

Mr. Durant's subject was "Rocket Power and Its Effect on Our Lives." In his address, he pointed up the impact of current research and development on rocket engines upon the general public in addition to its effect on national defense. Mr. Durant outlined the wide scope of related research under way, and how such work could be expected to affect our lives in the near and distant future.

Indiana Section Holds Election of 1953–1954 Officers

THE ARS Indiana Section held its annual business meeting on April 29, 1953, and elected officers and Board of

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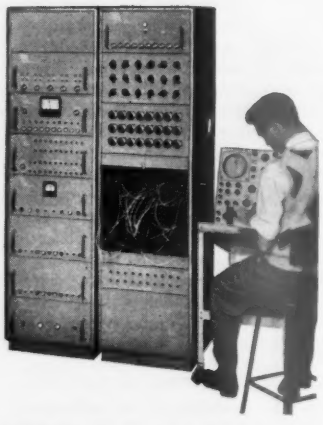

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For complete data, please request Bulletin F7.

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Directors for the year 1953–1954 as follows: *President,* Jacobus M. Botje: *Vice-President,* Alfred R. Graham; *Secretary,* Richard Foster; *Treasurer,* Richard E. Watters. *Board of Directors,* Dr. M. J. Zucrow, Faculty Adviser, Purdue University; Dr. W. L. Gilliland, Delbert E. Robison, Elliott L. Katz, and Wayne J. Colahan.

Von Braun Speaks at May Meeting

On May 6, the Indiana Section officially concluded a progressive 1952–1953 year with its Fourth Annual Banquet. Approximately 150 members and guests were present to hear the guest speaker, Wernher von Braun, chief of Technical Development Division, Redstone Arsenal, Huntsville, Ala. Dr. von Braun, who spoke on "Space Travel—How Soon?" stressed the urgent need for a well-organized and integrated program to study the problems related to the development of a satellite space station.

The Section announces that its membership over 1952–1953 has increased by 20 per cent. Six open meetings were conducted during that period, offering programs which covered the whole field of jet propulsion—missile development, space flight, nuclear power for aircraft, and ramjets. The Section is indebted to industry for providing so many exceptionally well-qualified speakers.

ARS 1952 Annual Report Available

THE ARS 1952 Annual Report has been completed and is being distributed to all ARS active members. Members of other classifications may obtain copies of the report by writing to the Secretary, American Rocket Society, 29 West 39th Street, New York 18, N. Y.

Southern California Section Tours Edwards Air Force Base

ONE hundred and twenty members of the ARS Southern California Section visited the Air Force Flight Test Center on May 22, 1953, for a tour of the Edwards AF Base facilities that were related to the rocket field.

The group was escorted by members of the AFFTC, with Major A. W. Thompson in charge. Arriving at the South Track, the members witnessed a hot test run of a rocket sled. At 11:00 a. m., they were escorted to the Rocket Test site. Mr. Gomperts, Chief of the Rocket Branch, gave a 20-minute lecture outlining the objectives of the Experimental Rocket Engine Test Station. The group was then shown a movie pertaining to various static firings of rocket propulsion power plants systems. After the movie, the members were escorted to the test area of the Test Station, which included major facilities such as the static firing test stand control station, and general machine shop.

The members returned to the Administration Building where refreshments were available before journeying back to Los Angeles.

Book Reviews

H. S. SEIFERT, California Institute of Technology, *Associate Editor*

ROCKET PROPULSION, by E. Burgess, Chapman and Hall, Ltd., London; The Macmillan Co., New York, N. Y., 1953, 235 pp. $4.50.

Reviewed by H. S. SEIFERT
California Institute of Technology
Jet Propulsion Laboratory

The author states in his preface to this interesting volume that he wishes to provide a link between popular and technical writings on rockets. Evidence of his fifteen-year connection with the British Interplanetary Society reveals itself in his broad perspective of the many problems related to rocketry and in his good qualitative insight into the physical processes of this field.

There are eight chapters, covering fundamental dynamics, fuels, motor design, feed systems and injectors, control, long-range missiles, extraterrestrial flight, and applications of nuclear energy. A notable effort is made to cope with the intricacies of propellant performance calculations and propellant evaluation in verbal terms.

This reviewer feels that the book will have value as "cultural" background in rocketry for students and persons with engineering training. The style remains lucid, although it is severely taxed by the effort to express in words concepts from thermodynamics, ballistics, reaction kinetics, and physical chemistry which are ordinarily mastered only in rigorous and quantitative engineering courses. The number of references in the book is small, being limited largely to British work of not very recent vintage.

The principal criticism which the reviewer feels is that the treatment is rather uneven, in some sections treating elementary matters in lengthy and discursive detail, and in others inserting mathematical material and obscure concepts without defining the symbols used. Thus, on page 65, entropy is introduced without comment, and on page 47 the chemical equilibrium equations are stated rather abruptly in *medias res*. A rather compressed derivation of the thrust and mass-flow equations is given in an appendix. On page 68 a curve of the influence of chamber pressure and mixture ratio on jet velocity is given with what seems to be a sparse explanation. A few mechanical flaws were noted, such as the statement on page 43 that the maximum theoretical specific impulse of aniline and nitric acid was 419 sec. Perhaps this was a misprint and was intended to be 219 sec. In this connection, it is felt that the definition of specific impulse given on page 19, as the "time which unit mass of fuel can support itself," would be more meaningfully stated as the "time during which unit weight of fuel can exert unit force," since "itself" is being uniformly dissipated.

In spite of the criticisms voiced, this reviewer feels that Mr. Burgess has made a worthy contribution to the scanty open literature of rocket propulsion. The documentation of his volume is limited and archaic, but his knowledge of the field, particularly of German work, is extensive, and his appreciation of its problems is up to date. It is a bold venture to assail thermodynamics with so blunt an instrument as the Queen's English.

ROYAL AERONAUTICAL SOCIETY MONOGRAPHS ON METALLIC AND OTHER MATERIALS, Vol. 1, The Properties of Metallic Materials at Low Temperatures, by P. L. Teed, Chapman and Hall, Ltd., London; John Wiley & Sons, Inc., New York, N. Y., 1950, 222 pp. $3.50.

Reviewed by HOWARD MARTENS
California Institute of Technology
Jet Propulsion Laboratory

This book, which is primarily a survey, will find its principal use as a reference for engineers or designers dealing with the problem of metal structures at low temperatures. In the author's own words, it is the result of "world-wide search for information" on the subject. The large amount of experimental data presented in the 45 tables makes this serve almost as a handbook. However, the explanatory text is a valuable addition to the figures given in the tables. The user will also find this book of great value because of the long list of references given at the end of each chapter. Each chapter which deals with a limited aspect of the subject also carries a conclusion in which the author points out additional factors to be considered when using the data presented.

The first chapter gives a brief review of the subject of metals at low temperature and of its relation to fundamental physics. One chapter deals with aluminum and its alloys. Ferritic and austenitic steels, because of their importance as engineering materials, receive consideration in three chapters. One chapter is devoted to magnesium alloys, and another to copper and its alloys. Nickel, zinc, tin, and lead and their alloys are treated together in one chapter

The user of this book will find that there are many lacunae in the material presented. However, this is not the fault of the author, for these deficiencies are in our present knowledge of the subject and merely point up the great need for reliable research in this important field.

RAKETENANTRIEBE, by J. Stemmer, Schweitzer Druck- und Verlagshaus A. G., Zurich, Switzerland, 1952, 523 pp., 206 illus., 27 tables. $5.

Reviewed by F. N. WOLF
California Institute of Technology
Jet Propulsion Laboratory

"Rocket-Propulsion Plants, Their Development, Application and Future," published in the German language in Switzerland, was written as an easily readable introduction into the field of rockets and space travel on a not-too-technical basis.

The author, a writer on subjects related to rockets for many years, collected information he received during the postwar years and compiled a quite extensive historical survey in the form of a chronological report on rocket developments from the mythological beginnings of Icarus' flight to the very latest missile performance reports. In six chapters he gives a general outline of the basic principles of the rocket motor in elementary form. The topics are fuels (solid and liquid), and the problems of high-speed flight, performance of the rocket motor, and its structural and functional parts. He draws on a manifold of sources of information and includes personal contributions by two German rocket experts on military rockets and structure.

A large section of the book is dedicated to an encyclopedic description of rockets and rocket-propelled airplanes that were actually built by the major contestants during World War II, and contains a wealth of details on construction and performance from the "Katiushka" of the eastern front to the V-2 missiles and the "Bat" in the west.

There are valuable tables on rocket fuels with their well-known code names as used by the Germans, and many novel ideas of European rocket engineers on design principles, such as characteristic length and nozzle configuration, chamber design as affected by unstable combustion, film cooling, and many more. A final chapter deals with the possibilities for space travel, artificial satellites, and their economic and astrophysical problems as well.

This book is well illustrated and contains a small bibliography which includes many European sources of literature in addition to the JOURNAL OF THE AMERICAN ROCKET SOCIETY and other familiar American publications.

MICROMETEOROLOGY, by O. G. Sutton, McGraw-Hill Book Co., New York, N. Y., 1953, 333 pp., $8.50.

Reviewed by E. BOLLAY
North American Weather Consultants
Pasadena, Calif.

The book deals with the fine structures of atmospheric processes which occur in the surface air layers. During recent years much emphasis has been paid to the physics of the atmosphere near the ground. This book makes available the results of this research.

Professor Sutton, who is an acknowledged leader in this field, has presented the material in detailed and precise exposition, bringing up to date the information on this subject previously covered in Brung's "Dynamic Meteorology."

This book consists of eight chapters. The earlier chapters deal with the physics and mathematics of the processes taking place near the earth's surface, and the

later chapters deal with applications of these concepts to the problems of wind structure near the surface, diffusion, and evaporation. The excellent bibliography at the end of each chapter will serve the graduate student in meteorology as well as the professional "applied meteorologist" in gaining access to source data. It starts with "The Atmosphere at Rest" in Chapter 1. Two chapters are devoted to "The Atmosphere in Motion," including both laminar and turbulent flow. Chapters 4 through 6 deal with "Heat Transfer and Problems of Diffusion," "Radiation," and "The Temperature Field in the Lowest Layers of the Atmosphere." Chapter 7, describing the "Problems of Wind Structure near the Surface," is one of the most comprehensive discussions of the subject which appears in a meteorological text in the English language. The last chapter on "Diffusion and Evaporation" is similarly complete and deals with Professor Sutton's own classical contribution to atmospheric diffusion.

This book is highly recommended to the graduate student and is essential for the professional meteorologist engaged in applied meteorology. It will be useful to the engineer and research scientist who has need to work in the surface layers of the atmosphere, particularly in the fields of diffusion and turbulence. It fills an important gap in the meteorological literature and will undoubtedly become a standard classroom text in professional meteorological courses.

PRINCIPLES AND METHODS OF SHEET-METAL FABRICATING, by George Sachs, Reinhold Publishing Corp., N. Y., 1951, 526 pp. $11.

Reviewed by R. E. MOULTON
California Institute of Technology
Jet Propulsion Laboratory

This book is an excellent summary of many modern sheet-metal forming techniques, and it endeavors to apply a scientific approach to the art of metal forming. The principles and methods of practically every common method of sheet-metal forming are discussed, and practically all available data are included; however, even with this mass of information and data, it still falls short of covering every specific case which one might encounter. Its best use would be that of a general guide in the selection of methods and procedures for a particular problem. It would also provide guidance for a logical approach to such a problem.

The field of metal forming is so broad and has such an infinite variety of applications and designs that no single volume or treatise can describe exact techniques to be used in all cases, and the publisher's review of this book is misleading in this respect. Given a design for a formed metal part, there can be a number of correct methods for producing this part which will vary according to the quantity of parts to be produced, the dimensional tolerances to be held, and most important, the material specified. In such an instance, this book will indicate certain methods for a specific material, but the questions of dimensional control, quantity production, and the difficulties introduced by alloys or materials other than those discussed are neglected. The volume would serve as an excellent reference for suggesting the approach to a particular metal-forming problem or as an excellent text for study in this field. It would also serve to furnish a specific answer to those few problems which are thoroughly discussed; however, it does not supply the cure for all metal-fabricating problems.

AERODYNAMICS OF PROPULSION, by D. Küchemann and J. Weber, McGraw-Hill Book Co., New York, N. Y., 1953, 340 pp. $9.

Reviewed by F. E. Marble
California Institute of Technology
Jet Propulsion Laboratory

Before the reader of this book mourns the brevity with which his own interests are dispatched, it is well to note the authors' modest claim to the effect that "... it cannot be expected to be exhaustive and it necessarily reflects the authors' preferences. But it is hoped that the subjects selected will fill some gaps in the published material available ...." This objective has been achieved and the authors merit the gratitude of engineers interested in the problems of power plant installation. In spite of the distinctly spotty character of the information, it will be recognized that those fields of propulsion aerodynamics to which Küchemann and Weber have contributed are discussed in a creditable manner. The book constitutes, furthermore, a sort of pioneering effort toward collecting this large group of fluid dynamic problems not considered part of the "classical" aerodynamic theory.

For purposes of review the material of the book may be subdivided into (1) Problems of air intakes, diffusers, nacelles, and discharge nozzles; (2) elements of the ducted propeller, ramjet, and turbojet power plants; and (3) aerodynamic problems associated with power plant installation, location, and accessories. The content of the first division centers about the authors' original work on subsonic inlets, cowlings, etc., and constitutes the most significant portion of the book. Although one might desire more detail on some points of the calculation procedure for special cases, this collection of material will be welcomed by those who have found occasion to refer to the original papers and monographs. Engineers engaged in the development of marine propulsion systems will find this work of interest in connection with ducted and submerged propeller installations. Since the mechanics of rotational fluid fields is not familiar to many students, an instructor using the book may find it necessary to provide introduction material on the subject.

The second division concerning jet power plants is unfortunately weak. In all fairness it should be allowed that the purpose of the discussion may be to orient the reader as to the relation between power plant performance and the problems of ducts and inlets. The treatment is confined, however, to elementary cycle analysis; the turbine and compressor elements are not treated and combustion chamber problems are introduced only formally. As a consequence the authors cannot discuss the difficulties encountered in component matching, a field in which diffuser performance can play a vital role. Throughout the book, the authors employ the pressure-volume and the temperature-entropy diagrams, devices which are not completely suitable for discussing jet engine performance. Furthermore, considerable valuable space (at 2.7 cents per page!) is devoted to evaluating engine "efficiencies" of rather dubious consequence.

The final portion is highlighted by information concerning cooling ducts, radiators, coolers, and their installation problems. Although a more thorough treatment of the heat transfer process would be highly desirable, the general aerodynamic problems are pointed out. Also included is a brief survey of the problems in handling the propulsive jet after discharge, and a discussion of aerodynamic propulsion in nature.

The book suffers, on the whole, by restriction to the subsonic field, but one may sympathize with the authors in their effort to collect unclassified information in the high-speed regime. Furthermore, discussion of real-fluid phenomena is notably absent. There are some minor items which may mislead a new student. For example, when the Mach number is first introduced, it is defined as the ratio of local fluid velocity to the free-stream sonic velocity—a convention peculiarly unsuited to the treatment of power plants.

These features do not, however, detract seriously from the worth of the book. It will be of value to students of propulsion, aeronautics, and mechanical engineering, and some of the material collected can find use as convenient reference for practicing engineers. The authors are to be congratulated on their worth-while effort.

FUNDAMENTAL PRINCIPLES OF POLYMERIZATION, by G. F. D'Alelio, John Wiley & Sons, Inc., New York, N. Y., 1952, 517 pp. $10.

Reviewed by S. Dobrin
California Institute of Technology
Jet Propulsion Laboratory

Dr. D'Alelio has organized in one volume the fundamental physical chemistry of polymers and polymerization. There is no special emphasis on those polymers used industrially, nor is there much reference to the naturally occurring high polymers.

The text covers chemical stoichiometry, reaction kinetics, thermodynamics, and statistical analysis of polymerization reactions. Structural features are related to physical properties and solution properties. A chapter is devoted to experimental techniques for determination of molecular weight, and another to thermal and chemical degradation of polymers.

The book is recommended to anyone whose interests touch the preparation of polymeric materials. However, it is not intended to be a reference book of applied technology but rather a textbook demonstrating fundamental principles of polymerization. It is well organized and clearly written and contains an excellent bibliography of the literature.

FATIGUE AND FRACTURE OF METALS, by W. M. Murray (editor), John Wiley & Sons, Inc., New York, N. Y., 1952, 313 pp. $6.

Reviewed by D. E. HUDSON
California Institute of Technology

Fourteen papers presented by well-known authorities at a 1950 Symposium at the Massachusetts Institute of Technology are here issued under the able editorship of Professor William M. Murray. These papers, which will be of great interest to anyone concerned with the relationships between materials and design, are mostly of the nature of reviews of the present status of the subject, with indications of important problems for future research.

Two main subjects are treated: first, the factors influencing the transition from ductile to brittle fracture in metals, under one or a small number of load applications; and, second, the ability of metals to resist a large number of repeated loads. In the first category are the following papers: M. Gensamer on the general significance of the transition temperature in mild steels; F. Jonassen on brittle fractures in ships; E. Orowan on the fundamental theory of brittle behavior in metals; P. A. Jacquet and A. R. Weill on temper brittleness in steel; and C. W. MacGregor on the influence of repeated loadings on the transition temperature. In the second category are the following papers: H. L. Dryden, R. V. Rhode, and P. Kuhn on a general consideration of the fatigue problem in aircraft structures; R. E. Peterson on the design of machine parts to resist fatigue; O. J. Horger and H. R. Neifert on the relations between internal stress and fatigue; R. L. Templin on some general factors in the design for fatigue; W. Weibull on some statistical features of the S-N curve; N. M. Newmark on cumulative damage in fatigue; P. L. Teed on the influence of metallographic structure on fatigue; N. J. Grant on fatigue at elevated temperatures; and J. T. Norton on the application of various techniques of physical metallurgy to the study of fatigue damage.

Useful bibliographies are given by all of the authors. Professor J. C. Hunsaker of the Massachusetts Institute of Technology has provided a foreword for the volume which explains the general background of the Conference.

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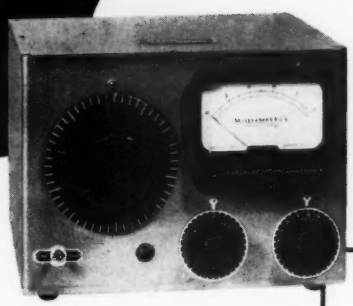

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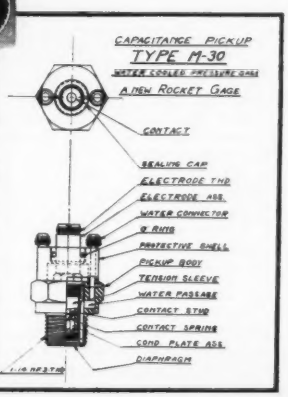

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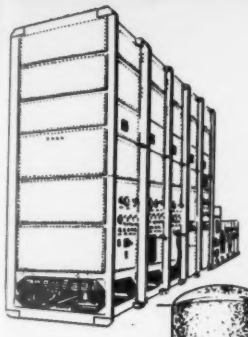

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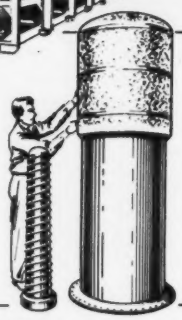

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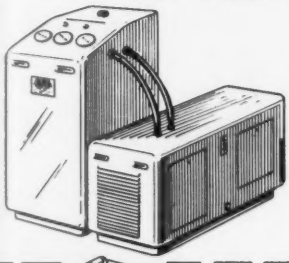

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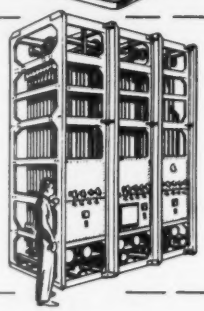

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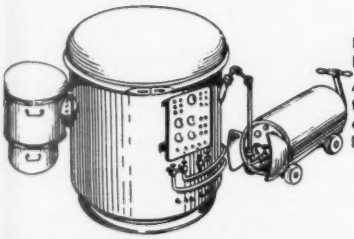

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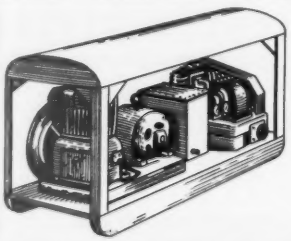

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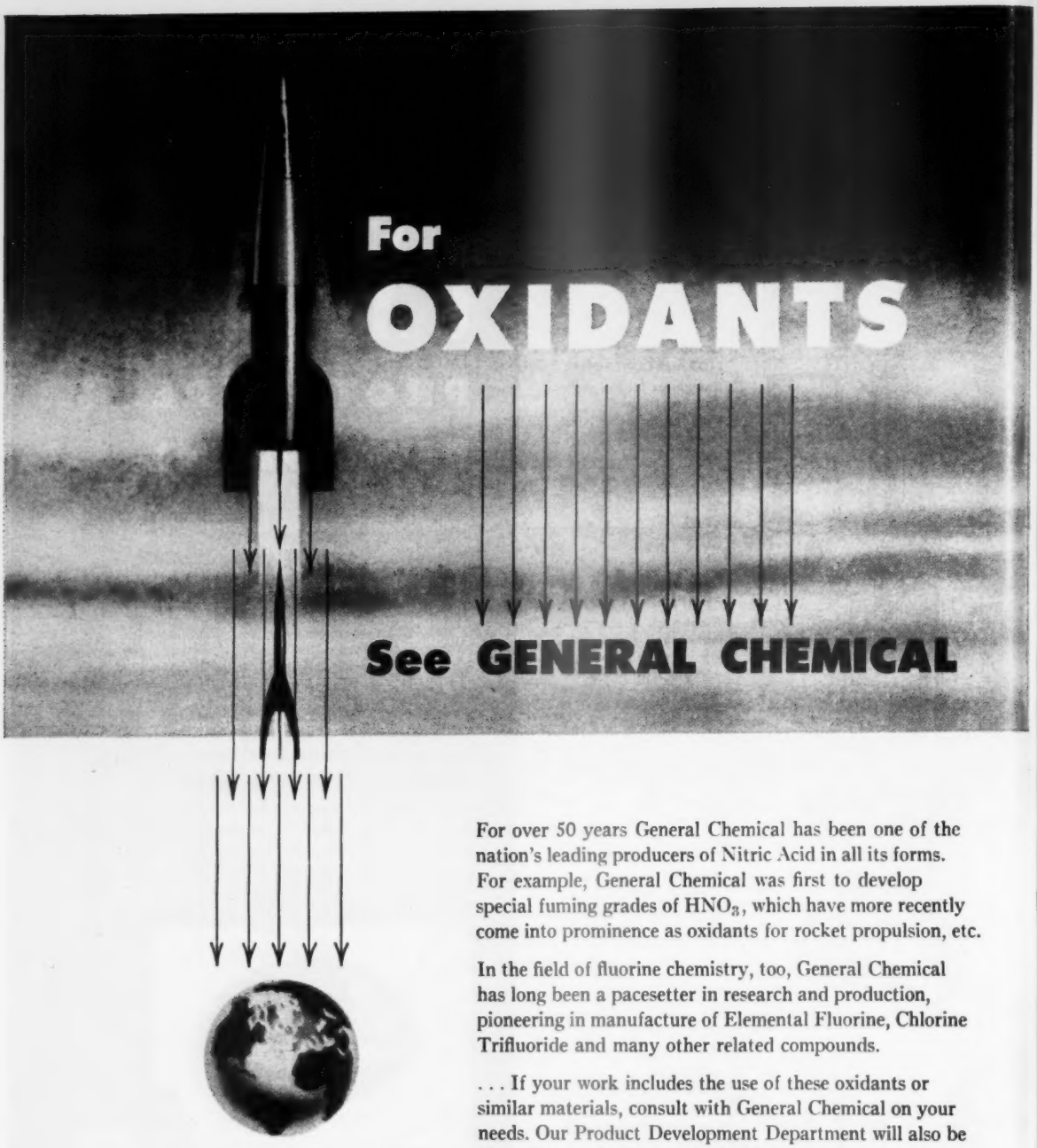

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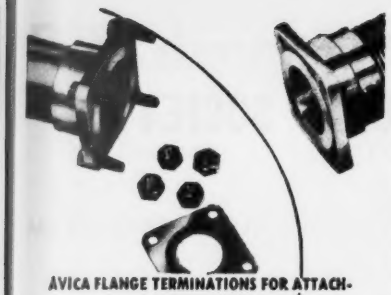

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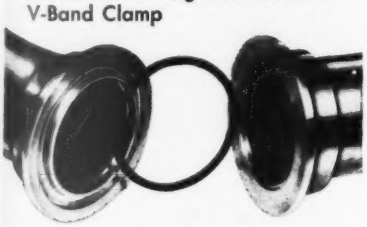

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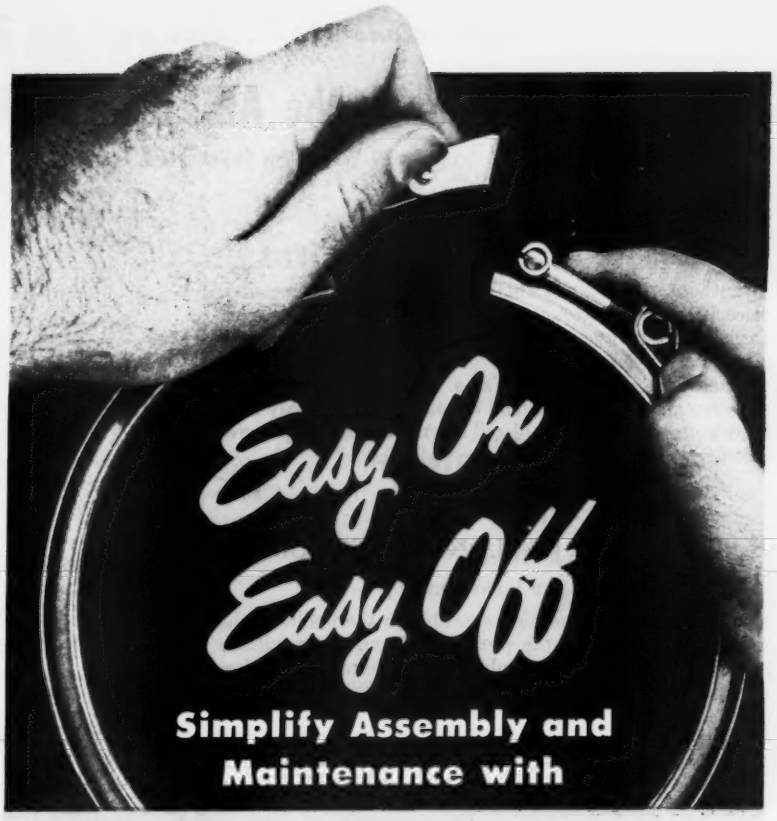

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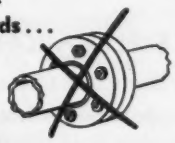

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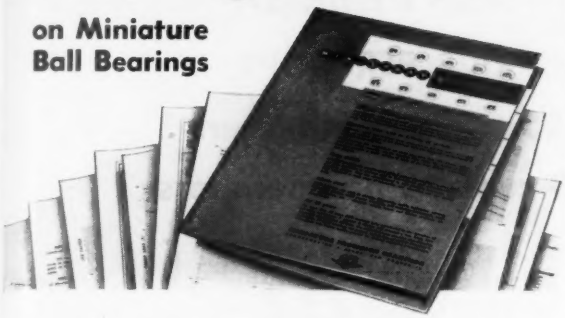

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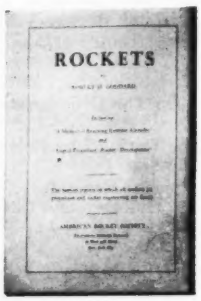

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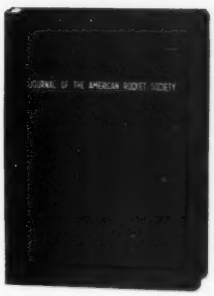

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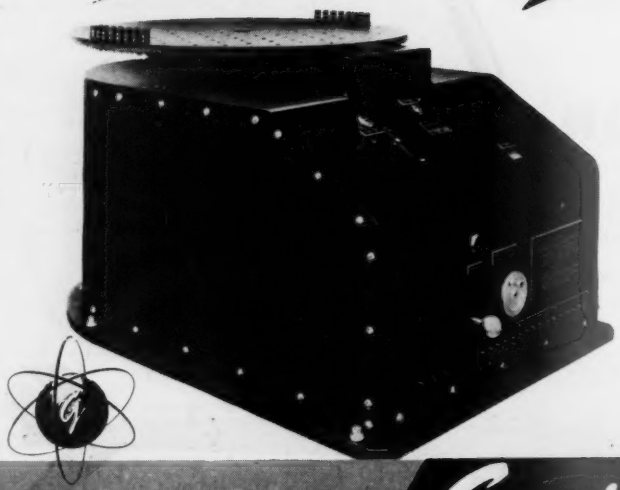

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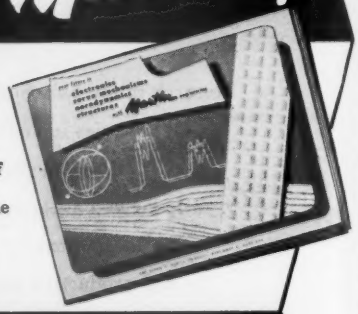

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7. Voltage Calibration: Signal amplitude is compared against a 1,000 cycle square wave (generated internally) the amplitude of which is controlled by a step-and-slide attenuator calibrated in peak volts. (A jack is provided to deliver 40V Peak for use in calibrating other instruments.)

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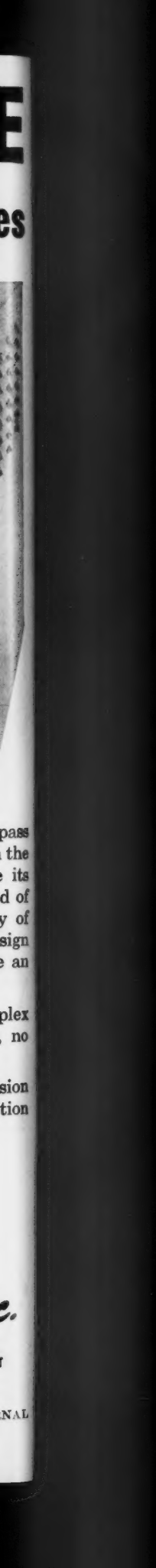
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